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RESEARCH MEMORANDUM

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ANALYTICAL COMPARISON OF TURBINE-BLADE COOLING SYSTEMS DESIGNED FOR A TURBOJET ENGINE OPERATING AT SUPERSONIC SPEED AND HIGH ALTITUDE

II - AIR-COOLING SYSTEMS

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SUMMARY

The influence of high-altitude supersonic flight on the operation and effectiveness of turbine-blade air- and liquid-cooling systems for turbojet application in guided missiles and supersonic interceptor aircraft were investigated analytically. The turbine blades in such applications must be effectively cooled to obtain the increased engine air-handling capacity, blade tip speed, and turbine-inlet gas temperature required for adequate specific engine thrust and weight. The problems encountered in air-cooling systems were investigated with reference to several specific designs for alternate heat-rejection mediums, and the results were compared with a similar study of liquid-cooling systems presented in a companion report of this analysis.

Results of the investigation showed that the simple, nonrefrigerated, air-cooling system was adequate for turbine-blade cooling at flight Mach numbers to 2.5 at an altitude of 50,000 feet and a turbine-inlet temperature of 2040° F. Air-cooling systems with bleed aftercooling or refrigeration based on heat rejection to afterburner fuel provide a greater margin of safety than the simple nonrefrigerated systems, extend operation to higher permissible flight Mach numbers without basic change in the engine, and provide the least mechanical complication and operating problems. Where limitations can be placed on the required cruising endurance, the most desirable alternate choice for supersonic interceptor and guided-missile engines is the fuel-cooling system using afterburner fuel, since it has minimum weight and performance penalties and is a relatively simple installation. The most promising applications for air- and liquid-cooling systems based on heat rejection to afterburner fuel are for missions in which the afterburner operates continuously throughout the flight, thus avoiding heat rejection to the main fuel tanks during nonafterburning operation. Regenerative liquid-cooling systems and regenerative air-bleed expansion refrigeration systems are promising long-range developments that are capable of operation over the desired range of flight conditions without affecting other aircraft installations and systems. Applicability of turbojet engines at supersonic flight speed can be extended with turbine-cooling systems that permit operation with increased turbine-inlet temperature, specific mass flow, and blade tip speed for flight Mach numbers to 2.5 at altitudes to 50,000 feet.

INTRODUCTION

Supersonic interceptor aircraft and guided missiles that are powered by turbojet engines may introduce turbine design requirements that will necessitate cooling of the gas-turbine components. As explained in reference 1, increased engine thrust and minimum specific engine weight will produce a trend towards increased turbine-blade tip speeds, longer turbine blades to provide for greater air-handling capacity, and higher turbine-inlet temperatures. These three conditions result in rotor-blade stresses and blade temperatures that are beyond the capacity of the best available high-temperature turbine-blade materials. A solution to the problem is to provide cooling to reduce the blade temperatures to a point where available turbine-blade materials will have sufficient strength to withstand the increased stresses that result from long blades and high tip speeds in improved engine designs. Methods of liquid-cooling turbine-rotor blades for application in a supersonic interceptor are compared in reference 1, and it is shown that possible disadvantages to all the systems are high heat-rejection rates due to over-cooling of the turbine blades and the problem of ultimate heat rejection to the available heat-rejection mediums. Supersonic flight also provides problems in the use of air cooling of turbine blades. The reduction in blade temperature that can be obtained with a simple air-cooling system is limited by the high compressor-bleed-air temperature and ram temperature associated with supersonic flight, although the effectiveness of the system can be augmented by bleed aftercooling or refrigeration which, in turn, requires rejection of heat. The problems encountered in either air or liquid cooling of turbine blades at supersonic flight speeds become more severe as the speed is increased and probably determine the ultimate limits of application of turbojet engines. For this reason, further studies of the application of cooling to the turbine blades are being undertaken.

Analytical and experimental research has already been conducted on application of air-cooled turbines to typical production turbojet engines to permit substitution of noncritical low-alloy-steel turbine blades and disks. Experimental investigations of heat-transfer characteristics and durability of turbine blades cooled with air under conditions equivalent to full-scale engine operation are reported in references 2 to 4. These and other investigations demonstrated that satisfactory cooling effectiveness could be provided with air-cooling methods to permit the substitution of low-alloy shells for currently used high-temperature alloys. Additional investigations of a complete air-cooled turbine rotor in a turbojet engine (references 5 to 7) have been conducted to explore full-scale application of air cooling where a complete system for supply and control of the coolant flow must be considered. Application of air-cooling systems to future high-performance engines to permit a simultaneous increase in turbine-inlet temperature, specific mass flow, and blade tip speed, however, is beyond the scope of present experience.

Therefore, predictions as to the suitability of various types of air-cooling system for turbojet engines at supersonic flight conditions are dependent upon analyses extending the available knowledge obtained at more conservative engine conditions.

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An analytical investigation was conducted at the NACA Lewis laboratory to evaluate the general cooling characteristics of various turbine air- and liquid-cooling systems and to compare the applicability of air- and liquid-cooled turbojet engines. Since an evaluation of turbine heat-transfer characteristics requires a knowledge of actual physical dimensions in addition to a thermodynamic analysis, a hypothetical fixed engine and aircraft design specification was chosen to obtain quantitative comparisons, from which a subsequent general comparison of air- and liquid-cooling systems was made.

The operating conditions imposed in the analysis were based on an assumed supersonic interceptor flight plan over a range of flight conditions to a Mach number of 2.5 at an altitude of 50,000 feet, and the cooled turbine design analysis was made for an assumed basic engine of a type and size appropriate for the gross weight and power loading of a supersonic interceptor aircraft. The basic engine design specifications, which represent a considerable advance in some respects over current-production jet engines, include a sea-level specific mass flow of 23.6 pounds per second per square foot, a turbine-inlet temperature of 2040° F, a sea-level compressor pressure ratio of 6.0, a turbine tip speed of 1500 feet per second, a turbine diameter of 35.1 inches, and a turbine hub-tip ratio of 0.732.

Specific objectives of this air-cooling-system analysis were essentially the same as for the liquid-cooling-systems analysis of reference 1 and are as follows:

(1) Determination of the capacity to adequately control rotor blade temperature within design limits under the specified engine and flight conditions

(2) Determination of which flight condition represents the most critical cooling requirement

(3) Determination of the limitations imposed on rotor blade cooling effectiveness by bleed-air temperatures and heat-rejection mediums used for aftercooling or refrigeration cycles

(4) Evaluation of the modifications that could be made to the cooling cycle to avoid limitations imposed by the heat-rejection mediums

(5) Evaluation of the thermodynamic possibilities for completely self-contained, regenerative, air-cooling systems independent of engine

installation and auxiliary systems and rejecting heat to the main-engine working fluid at compressor-outlet conditions.

In this analysis no consideration was given to cooling of the stator blades.

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DESCRIPTION OF AIR-COOLING SYSTEMS INVESTIGATED

In all the air-cooling systems considered in this analysis, the cooling air is bled from the main-engine compressor at outlet conditions and, after cooling the turbine blade, the air is discharged into the main-turbine exhaust from the blade tip. The use of air at the compressor-outlet pressure is not necessarily required because of a need of high-pressure drop in the cooling system, but may be dictated by the compressor configuration. If sufficiently high blade tip speeds are used in the compressor design, it becomes possible to achieve the desired pressure ratio in a single compressor stage, thus making it impossible to obtain bleed air for cooling at an intermediate pressure. The high compressor-outlet temperature encountered at high flight Mach number, even with moderate compressor pressure ratio, may seriously limit the effectiveness of an air-cooling system unless means are provided for aftercooling or refrigeration of the cooling air before introduction into the turbine. Although different in function, heat rejection with air-cooling systems presents the same problems as with liquid-cooling systems (reference 1) and, therefore, special adaptations are required to meet conditions found in different engine installations.

Simple Air-Cooling System

In the simple air-cooling system (fig. 1), a small amount of cooling air is bled from the main-engine compressor at outlet conditions and introduced into the turbine-rotor disk at a point near the shaft axis. The coolant is then pumped out through the hollow disk passages and blades and is discharged from the blade tips. The cooling air then mixes with the main-engine gas flow and passes into the jet nozzle where it contributes to the engine thrust. The coolant temperature entering the blade base is equal to the main-engine compressor-outlet temperature plus allowances for heat transfer and additional compression as the coolant passes through the disks. The coolant flow is regulated by a throttling valve in the bleed line.

The simple air-cooling system appears to represent the minimum mechanical complication in the engine installation since the bleed-air line and control valve comprise the only additional equipment. The system is independent of the heat-rejection problems outlined for the liquid-cooling systems in reference 1, although, as the compressor-outlet

temperature approaches the desired turbine-blade temperature under unusual atmospheric conditions or high flight Mach number, excessive coolant-flow requirements may be encountered or the reliability of the turbine compromised through over-temperature operation. A disadvantage of the simple system is, therefore, that coolant temperature, which is equally as important as coolant flow, is left to chance and subject to random flight and atmospheric variations. Addition of bleed aftercooling or refrigeration systems is therefore essential to extend the flight Mach
2758 number capabilities of the turbojet engine, to minimize the coolant-flow requirements, and to provide protection against inadvertent over-temperature operation.

Bleed with Water Injection

A method of bleed aftercooling that can be incorporated in the simple air-cooling system illustrated in figure 1 with little mechanical complication is water injection into the coolant downstream of the throttling valve. Injection of a relatively small quantity of water at this point in the system accomplishes a considerable reduction in air temperature up to the point of saturation of the air and, if desired, excess water can be entrained in the air for subsequent evaporation within the blade coolant passage. Application of this system is dependent upon the flight duration since the water carried by the aircraft for this purpose is lost throughout the mission.

Bleed-Aftercooling System

A bleed-aftercooling system with heat rejection to afterburner fuel is illustrated in figure 2(a). The simple air-cooling system in figure 1 was modified by addition of an aftercooler in the bleed line upstream of the throttle valve; in the aftercooler the coolant temperature is reduced to a specified value by heat rejection to the afterburner fuel as it is drawn from the fuel tanks. The coolant temperature reduction is determined by the size and the effectiveness of the heat exchanger and the fuel temperature, since ample heat capacity is available in the afterburner fuel flow with the heat-rejection rates in this case being considerably smaller than in the fuel-cooled turbine applications (reference 1). The aftercooling requirements for nonafterburning engine operation are provided by recirculating fuel to the fuel tanks, as previously described in reference 1 for the fuel-cooled turbine except that the conditions are less severe. A bleed-aftercooling system with heat rejection to ram air is an alternate method for consideration, but was not investigated in this analysis because of the undesirable ram-air inlet ducting and exhaust disposal problem in a nacelle engine installation. As with the water-cooling system (reference 1), large amounts of air from engine-inlet boundary-layer-removal slots may be available in a fuselage engine installation that would serve for bleed aftercooling.

Air-Bleed Expansion Refrigeration Systems

A considerable pressure difference may exist between the compressor bleed point and the point of introduction of the coolant into the turbine rotor, if it is assumed that it is undesirable or impossible to bleed the compressor at an intermediate point because of its configuration. A further temperature drop can therefore be obtained by replacing the throttling valve with an expansion turbine. An expansion refrigeration system for bleed aftercooling is illustrated in figure 2(b), where an auxiliary expansion turbine is placed in the bleed line following the aftercooler. The pressure ratio across this turbine is generally about the same as that across the main turbine. The work produced by the expansion turbine as a part of the refrigeration process must be absorbed in some manner and is available for operation of other accessories such as afterburner fuel pumps or possibly for pumping of afterburner-shroud cooling air. An increased expansion ratio across the refrigeration turbine can be achieved in the system illustrated in figure 2(c), where the power of the auxiliary turbine is used to drive an auxiliary compressor in series and ahead of the aftercooler. This modification represents an additional complication, but increases the refrigerating capacity of the auxiliary turbine. The gain relative to the system illustrated in figure 2(b) is dependent upon the effectiveness of the aftercooler. The bleed aftercooling and expansion refrigeration systems illustrated so far utilize the afterburner fuel and complicate the engine installation because they both depend upon and influence the plumbing of the afterburner fuel system. The arrangement illustrated in figure 2(c) can be modified, however, to avoid this limitation and provide as well for nonafterburning turbojet engines where no readily available heat-rejection medium such as the afterburner fuel flow is available.

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Regenerative Air-Bleed Expansion Refrigeration System

The regenerative expansion refrigeration system is illustrated in figure 2(d), where the aftercooler between the auxiliary compressor and expansion turbine is relocated at the discharge of the main-engine compressor. In this system the cooling air bled at compressor-outlet temperature is compressed further in the auxiliary compressor in order to increase the temperature to a point where heat can be rejected back to the main-engine working fluid. The cooling air emerges from the aftercooler at a temperature somewhat higher than main compressor-outlet temperature but at a considerably higher pressure. It is then possible to refrigerate the cooling air by expansion through the auxiliary turbine.

Thus, an appropriate arrangement of the auxiliary components of the bleed aftercooling and refrigeration system can be made that results in a self-contained or "package" installation for the air-cooled engine if the design conditions imposed in the interceptor-aircraft engine application warrant the additional refinement.

ANALYTICAL METHODS FOR EVALUATING AIR-COOLING SYSTEMS

The general analytical procedure followed is similar to that presented in references 8 and 9, where the turbine configuration and the engine operating conditions over a range of altitude and flight speed are used to define the heat-transfer rates, the coolant temperature, the required coolant flow, and the blade temperature, which are the most essential factors in the evaluation.

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Design Criterions

Compressor-bleed air is less effective than liquids for turbine-blade cooling, and application under the conditions encountered in the interceptor-aircraft mission of this analysis illustrates the severe specifications that must be fulfilled. Because the cooling air is taken from the main-engine working fluid and passed only once through the turbine-blade coolant passage, all possible effective steps are generally taken to minimize the coolant-flow rate. The two most significant variables are the coolant flow and the coolant temperature. The characteristic behavior of air cooling is that the greatest blade temperature reduction is achieved with the first increments in coolant flow, with rapid onset of diminishing returns. The addition of extended internal heat-transfer surfaces in a blade tends to reduce the general level of coolant flow required, but the diminishing-returns characteristic becomes more pronounced. It is largely as a result of this behavior that coolant temperature becomes so significant since, with a specified blade temperature and high coolant temperature, the point of diminishing returns can easily be exceeded and even extreme coolant-flow rates will not adequately control blade temperature. Under these circumstances, regardless of coolant-flow rate, there can be no margin of safety in the design. Two alternatives remain open to circumvent this limitation as design conditions become more severe. The first is a major redesign of the blade configuration to further increase the general level of cooling effectiveness; the second is to introduce special methods for control of coolant temperature. The first approach may lead to costly new development and fabrication problems in any given engine since the entire turbine design may be affected. The second approach permits retention of a fully developed air-cooled turbine-rotor design and confines the new developments to external components that can be isolated from the main engine.

The first design criterion usually considered in an air-cooling system is the permissible coolant-flow rate. For nonafterburning turbojet engines of the type and size considered in this analysis, a coolant flow of approximately 2 to 3 percent of the main-engine mass flow would be considered reasonable since the appreciable gain in engine thrust resulting from high turbine-inlet temperature would not be greatly affected by

the cooling losses. Cycle analysis of air-cooled afterburning engines of the type and size considered in this analysis indicates that the cooling losses are negligible, and consequently the coolant-flow rate for afterburning engines does not appear to be a critical criterion from the standpoint of over-all engine performance.

Many other factors, however, necessitate minimum coolant-flow rates and are intimately related to detailed engine design problems. The pressure drops associated with high coolant-flow rates may exceed the capabilities of the bleed system at some portion of the aircraft mission. The facilities required for introduction of the coolant into the rotor may be simplified where small volume flow is required. In addition, very little is known about the influence of coolant discharge on the aerodynamic losses of the blading. With this perspective in mind, the principal design criterions for air cooling resolve into a balance between over-all blade cooling effectiveness, diminishing-returns characteristics, and control of coolant temperature. In the presence of diminishing returns, control of coolant temperature becomes a major factor in achieving a margin of safety in the blade. The best over-all criterion for an air-cooling system is the margin of safety and the ability of the system to accommodate increasingly severe design conditions. A system operating under marginal conditions would be considered less desirable for extensive research and development. 2758

Since the essential systems for aftercooling and refrigeration of the blade cooling air previously described all involve rejection of heat, an additional criterion is the ability of the system to function with whatever heat-rejection mediums are available, either ram air, afterburner fuel, or the main-engine working fluid. Secondary design criterions of the bleed aftercooling and refrigeration systems are the probable weight and degree of mechanical complication. The manner in which these factors are related in evaluating air-cooling systems is indicated in subsequent sections.

In order to evaluate air-cooling systems analytically, it is first necessary to determine the variation of blade-profile heat-transfer coefficients over the range of altitude and flight speed in the same manner as for liquid cooling (reference 1), and with an established coolant passage configuration it is necessary to determine the relations between blade cooling effectiveness and the coolant-flow rate for the particular engine operating conditions of interest. Then, with consideration of the coolant bleed temperatures, the strength properties of the blade material, and the stresses encountered, it is possible to evaluate a required coolant flow. Further analysis can then be made of the balance between coolant-flow rate, coolant temperature, and the margin of safety at critical points in the flight plan, with consideration of known endurance characteristics of air-cooled blade configurations. Where the general analysis indicates that bleed aftercooling or refrigeration is necessary,

additional cycle calculations may be made to determine the reductions in coolant temperature available through special adaptations of the system and, where possible, the probable sizes and weights. The limitations encountered in rejecting heat from coolant refrigeration systems are similar to those previously described for liquid-cooling systems (reference 1), except that special consideration is given in this analysis to application of expansion refrigeration units rejecting heat to the afterburner fuel or the main-engine working fluid.

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Equation for Average Blade Temperature

A one-dimensional solution of the spanwise temperature distribution in a turbine blade is given in reference 10. The equation in reduced form is as follows:

$$\varphi = \frac{T_{g,e} - T_B}{T_{g,e} - T_{a,e,h}} = \frac{1}{1 + \lambda} e^{-\left(\frac{1}{1+\lambda} \frac{H_o l_o b}{c_{p,a} w_a} \frac{x}{b}\right)} \tag{1}$$

where

φ temperature-difference ratio

T_B average blade temperature

$T_{a,e,h}$ cooling-air temperature at blade root

$\lambda = \frac{H_o l_o}{H_f l_i}$

x/b position where T_B is obtained in terms of span $\left(\frac{x}{b} = \frac{1}{3}\text{ herein}\right)$

Reference 10 indicates that the two terms omitted from the original complete equation to get the form given above are of minor importance. Inasmuch as the calculations were made for the critical spanwise station, $x/b = 1/3$, and the blade geometry remained fixed, the significant parameters affecting the ratio φ were the gas-to-blade coefficient H_o (as calculated in reference 1) and the coolant weight flow w_a, which also determined the effective blade-to-coolant coefficient for an extended heat-transfer surface H_f as explained in reference 11 and the following section.

Gas-to-Blade and Blade-to-Coolant Heat-Transfer Coefficients

Both gas-to-blade and blade-to-coolant coefficients were required to obtain the required coolant-flow rates. The gas-to-blade coefficients were determined in exactly the same manner as described in reference 1.

The heat-transfer correlation equation used for evaluating the average convective blade-to-coolant heat-transfer coefficient $H_{i,B}$ can 2758
be obtained from equation (90) of appendix F of reference 12 and is, in the present notation,

$$H_{i,B} = 0.019(Re_{a,B})^{0.8} \left(\frac{k_{a,B}}{D}\right) \tag{2}$$

where

$$Re_{a,B} = \frac{w_a D}{A\mu_{a,B} g} \frac{T_a}{T_B} \tag{3}$$

The fluid properties in equations (2) and (3) are based on the blade wall temperature because it is shown in reference 13 that, for air flowing in a smooth round tube and for the range of wall and air temperatures normally encountered in air-cooled turbine blades, a satisfactory correlation is obtained for average inside heat-transfer coefficients when the fluid properties are based on wall temperatures.

The method for obtaining the effective heat-transfer coefficient H_f for a finned or corrugated type of surface when the convective coefficient $H_{i,B}$ is known is given in reference 11.

Method for Obtaining Required Coolant Flow

The critical cross section of an air-cooled blade generally occurs in the region from one third to one half of the span from the root. That is, failure is more likely to occur in this region because the combination of stress and temperature tends to make the blade weakest at this section. The one-third-span station was selected as representative of the critical section for this analysis, and all blade temperature calculations were therefore made for this station.

A relation between blade strength and blade temperature can be obtained from the centrifugal stresses acting on the blade, the stress-rupture relation between stress and temperature for the material, and the stress-ratio factor, which is explained in references 4 and 11 and defined as the ratio of the integrated allowable strength to the

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centrifugal stress. The stress-ratio factor is always chosen greater than 1 because the design integrated allowable stress of the blade must be greater than the centrifugal stress in order to make allowance for unknown thermal and vibratory stresses which cannot be evaluated accurately. Evaluation of the proper stress-ratio factor is dependent upon endurance tests of a particular blade in an engine. The method of evaluation and the experimental values of stress-ratio factor for an air-cooled blade with extended internal heat-transfer surface are given in reference 4. The stress-ratio factor obtained for a given blade and operating condition depends upon the blade temperature and therefore the cooling-air flow rate. On the basis of the results of reference 4, a design stress-ratio factor of 1.9 was chosen for most of the air-cooling analysis of this investigation.

The allowable average blade temperature can be determined by using the stress-rupture properties of the blade material and the calculated metal strength required, based on the centrifugal stress and the stress-ratio factor. The coolant flow required to obtain the allowable average blade temperature can be calculated by the procedure explained in the section Equation for Average Blade Temperature, which gives the relation between coolant flow and blade temperature for a given blade-coolant-passage configuration and for given values of effective gas temperature, outside heat-transfer coefficient, and cooling-air temperature.

Aftercooler Size Determination

Because no data were available on the characteristics of fuel-to-air heat exchangers and, since the heat-transfer properties of fuel and ethylene glycol are similar, the data for ethylene-glycol radiators were used to estimate the size and pressure drop of the cooling-air aftercoolers. The results represent first approximations only and are presented to indicate the order of magnitude of the aftercooler requirements.

The aftercooler characteristics can be obtained from a curve such as in figure 3, in which the heat-transfer rate in Btu/(sec)(sq ft of frontal area) per 100° F temperature difference between mean coolant and entering-air temperature and the corrected static-pressure drop through the core $\sigma_5 \Delta p_{5-6}$ are given for core lengths of 9 and 12 inches.

With the cooling-air flow rate and the desired cooling-air temperature reduction known, the approximate aftercooler size and pressure drop were estimated in the following manner: The face, or frontal area of the aftercooler, must first be assumed and then the core length is estimated by extrapolation or interpolation from the lines for 9- and 12-inch core lengths. The static-pressure drop of the air through the core is obtained by dividing the corrected pressure drop $\sigma_5 \Delta p_{5-6}$ by the correction factor

σ_5, which is defined as the ratio of the density at the cooler entrance to the standard sea-level density.

Cooling-Air Refrigeration by Water Injection

The temperature of the cooling air can be reduced to its saturation temperature by injecting water into the cooling-air duct. The saturation temperature of the cooling air at the various flight conditions and the water-air ratio required can be determined from a psychometric chart given in reference 14.

Calculation Methods for Refrigerating Cooling-Air Systems Using Auxiliary Components

As described previously, in the section DESCRIPTION OF AIR-COOLING SYSTEMS INVESTIGATED, systems for refrigerating the cooling air were explored in which first an aftercooler and an expansion turbine were used and second an auxiliary compressor, an aftercooler, and an expansion turbine were used. In the first case, the pressure drop across the aftercooler was neglected and the expansion occurred from compressor-outlet pressure to required cooling-air pressure at the engine turbine hub, which was assumed to be equal to the engine turbine-outlet pressure. The enthalpy drop across the auxiliary turbine is then

$$(\Delta h'_a)_{7-8} = \eta_t c_{p,a,7} T'_{a,7}\left[1 - \left(\frac{p'_{a,8}}{p'_{a,7}}\right)^{\frac{\gamma_a - 1}{\gamma_a}}\right] \tag{4}$$

where $T'_{a,7}$ is in $^{\circ}R$. From this enthalpy drop and the temperature at the aftercooler outlet, the cooling-air temperature drop across the expansion turbine and the cooling-air supply temperature to the engine turbine are readily calculated.

If the power obtained from the auxiliary turbine is used to drive an auxiliary compressor (see fig. 2(c)), the enthalpy rise in the auxiliary compressor is equal to the enthalpy drop across the turbine. Or

$$(\Delta h'_a)_{3-4} = (\Delta h'_a)_{7-8} = \frac{c_{p,a,3} T'_{a,3}}{\eta_c}\left[\left(\frac{p'_{a,4}}{p'_{a,3}}\right)^{\frac{\gamma_a - 1}{\gamma_a}} - 1\right] \tag{5}$$

The temperature at the inlet to the auxiliary turbine can be obtained on the basis of equations (4) and (5), and the resulting equation is

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$$T'_{a,7} = \frac{c_{p,a,3}T'_{a,3}\left[\left(\frac{p'_{a,4}}{p'_{a,3}}\right)^{\frac{\gamma_a-1}{\gamma_a}} - 1\right]}{c_{p,a,7}\left[1 - \left(\frac{p'_{a,8}}{p'_{a,7}}\right)^{\frac{\gamma_a-1}{\gamma_a}}\right]\eta_t\eta_c} \tag{6}$$

Inasmuch as it was assumed that no pressure drop occurred across the aftercooler, $p'_{a,4}$ equals $p'_{a,7}$. For given auxiliary compressor-inlet conditions $p'_{a,3}$ and $T'_{a,3}$ and given cooling-air supply pressure $p'_{a,9}$ ($p'_{a,9} = p'_{a,8}$), equation (6) can be solved for the auxiliary turbine-inlet temperatures as the pressure ratio $p'_{a,4}/p'_{a,3}$ across the auxiliary compressor is varied. The aftercooler must then provide the required amount of temperature reduction $(T_{a,5} - T_{a,6}) = (T_{a,4} - T_{a,7})$ in order to give the required auxiliary turbine-inlet temperature; or, in other words, after the pressure ratio of the auxiliary compressor and thus the auxiliary turbine-inlet temperature are established, the cooler size and capacity requirements are also established. After the auxiliary turbine-inlet temperature is known, the enthalpy drop across the auxiliary turbine is calculated from equation (4) and then the coolant-supply temperature can be calculated.

If the aftercooler size and capacity are specified, for given values of auxiliary compressor-inlet conditions, bleed-air-flow rate, aftercooler coolant temperature, and auxiliary turbine-outlet pressure, the auxiliary compressor, aftercooler, auxiliary turbine system balances out to the extent that only one auxiliary compressor pressure ratio can be used. Thus, there is one fixed cooling-air supply temperature for the condition of a given aftercooler size and capacity.

ENGINE DESIGN OPERATING CONDITIONS

The flight conditions, the basic engine specification, the aircraft specifications, the turbine specifications, the assumptions used in the air-cooled-systems evaluations, and the engine operating conditions for this investigation are summarized in tables I and II. The turbojet engine considered was designed to drive a twin-engine supersonic interceptor aircraft over a range of Mach numbers from 0 to 2.5 with primary

emphasis given to a combat Mach number of 1.8 at an altitude of 50,000 feet and sufficient engine power for a 2g maneuver at these conditions. The details of the flight plan and the engine specifications and requirements are given in reference 1. 2758

The detail specifications of the turbine are given in table I, and the air-cooled turbine-blade midspan configuration used in this analysis is shown in figure 4. This blade was not designed specifically for the turbine used in this analysis, but based on the required velocity diagram for driving the compressor it was found that such a blade would approximately fulfill the design conditions. The blade is the same as configuration H, profile 2, in reference 11. The spanwise distribution of centrifugal stress is shown in figure 5 for this blade for an area ratio of 0.5. Area ratio is defined as the ratio of metal cross-sectional area at the tip to the metal cross-sectional area at the root. At the one-third-span position the centrifugal stress is 33,500 pounds per square inch as compared with approximately 23,500 pounds per square inch in the J33 turbine used for the experimental investigations of references 2 to 7. The stress at the blade root is 44,500 pounds per square inch.

CALCULATION PROCEDURE

In order to evaluate the air-cooling systems, the methods of analysis described in the section ANALYTICAL METHODS FOR EVALUATING AIR-COOLING SYSTEMS were applied first to evaluate the range of coolant-passage heat-transfer coefficients obtainable with the extended internal heat-transfer surfaces illustrated in figure 4 for the air-cooled blade selected for the basic engine. Both average inside heat-transfer coefficients and the average effective coefficients of the equivalent finned surfaces were calculated with the use of equation (2) and reference 11 for a range of coolant flows to about $3\frac{1}{2}$ percent of the engine mass flow at sea-level static conditions.

In order to determine the characteristics of the nonrefrigerated system with direct compressor-bleed air, calculations were made over the range of flight conditions from sea-level static to a Mach number of 2.5 at an altitude of 50,000 feet. Required coolant-flow ratios were determined for a stress-ratio factor of 1.9, as explained in the section Method for Obtaining Required Coolant Flow, and for allowable blade temperatures based upon the 100-hour stress-rupture properties of the high-temperature alloy S-816. In all the air-cooling calculations with the nonrefrigerated system, the temperature of the cooling air entering the blade base was taken as that at compressor-outlet plus an allowance of 60° F for the temperature rise from the hub of the rotor disk out to the blade base. The operating conditions and specifications for this part of the analysis are given in tables I and II.

The influence of the design stress-ratio factor on the required coolant flow was investigated over a range of stress-ratio factor from 1.35 to 2.35 at the combat conditions of Mach number 1.8 and altitude of 50,000 feet to determine the sensitivity of the design and the probable margin of safety that could be provided within the desired range of coolant flow for this particular blade configuration. Larger stress-ratio factors will provide a greater margin of safety in a blade design; therefore, the purpose of this calculation was to evaluate whether the design stress-ratio factor could be increased to meet more severe
2758 requirements. If the stress-ratio-factor increase obtainable at higher coolant flow was inadequate, it would indicate that the point of diminishing returns had been exceeded and that an alternate blade configuration should be considered.

An additional question that always arises in cooled-turbine design is the number and the size of rotor blades to perform the required work with minimum cooling losses. It would be expected, for a given size of blade, that a low-solidity rotor would have lower cooling requirements because of the reduced number of blades and heat-transfer area, although it might become difficult to obtain the required work efficiently. Calculations were made at the combat Mach number of 1.8 and altitude of 50,000 feet to determine the variation in required design coolant flow, with consideration given to the changes in profile heat-transfer coefficients and coolant-flow areas over a range of blade chord and solidity that result in a minimum of 50 and a maximum of 150 blades on the rotor. All other conditions remained the same as those given in tables I and II for the nonrefrigerated system.

In order to investigate the influence of coolant temperature on coolant-flow requirements, the coolant temperature was varied from 700° to 100° F for the combat flight conditions. All other conditions remained the same as in tables I and II for the nonrefrigerated system. An additional design criterion investigated was the influence of coolant temperature reduction on the stress-ratio factor of this particular blade configuration in order to evaluate the margin of safety that could be realized in this manner through application of bleed aftercooling or refrigeration.

The effectiveness of various methods for reducing the coolant-supply temperature to the rotor was investigated by means of the procedures previously outlined. The heat-transfer characteristics of a system using water injection into the cooling air were determined for the take-off, climb, and combat portions of the mission to evaluate the water-to-air ratios, the weight of water required for each phase of the mission, and the influence of water injection on the required amount of compressor-bleed air. In each instance, the compressor-bleed air was reduced to the saturation temperature for the pressure level that prevailed, and the total weight of water required for the flight was evaluated as a design

criterion for comparison with the weight of a bleed aftercooler with heat rejection to fuel that achieves the same temperature reduction. In the latter case, for the system illustrated in figure 2(a), the pertinent design criterions such as heat-rejection rates and fuel temperature rise were evaluated over the range of flight conditions from sea-level static to the combat Mach number and altitude so that a critical design point could be selected. The size and weight of the aftercooler was then evaluated and its performance at other significant points in the mission checked. 2758

The characteristics of expansion refrigeration air-cooling systems were investigated at the combat Mach number and altitude for a fixed coolant-supply pressure to the turbine-rotor hub and a range of aftercooler-outlet temperatures from 100° to 900° F. The aftercooler-outlet temperature was used as the design criterion because of its significance when the heat-rejection medium available is considered and also because it dictates the auxiliary pressure ratio of the refrigeration cycle when the auxiliary compressor and the auxiliary turbine are coupled. Insufficient data prevented evaluation of the weights of these systems.

RESULTS AND DISCUSSION

Coolant Passage Heat-Transfer Coefficients

The coolant-passage heat-transfer coefficients obtainable with air for the blade configuration illustrated in figure 4 are given in figure 6 for a blade temperature of 1210° F and an average coolant temperature of 800° F for a range of coolant flows to about $3\frac{1}{2}$ percent of the engine mass flow at sea-level static conditions. The air-cooled blade considered in this analysis is a thin hollow shell fitted with a corrugated-fin insert that confines the coolant flow to the passages near the shell and augments the internal heat-transfer surface. The lower line gives the average convective coefficient from metal to air based on a heat-transfer correlation in which the fluid properties are evaluated at the passage wall temperature, as explained in the section Gas-to-Blade and Blade-to-Coolant Heat-Transfer Coefficients. The upper line gives the effective heat-transfer coefficient over the inside surface of the blade shell that results from the heat transferred through the equivalent finned surfaces provided by the corrugated-fin insert and the heat transferred directly from the inner surface of the shell. The effective coefficient represents the design values used in the engine analysis. At a coolant flow per blade of 0.05 pound per second, which would be typical for the engine considered, the convective heat-transfer coefficient is 77 Btu/(hr)(sq ft)(°F), but the addition of internal surface provides an

effective coefficient of about 225 Btu/(hr)(sq ft)(°F). The effectiveness of the fins is influenced by the temperature drop along the conduction path and is dependent upon the proportions of thickness and spacing and the thermal conductivity of the fin material. For ideal conditions, the ratio of effective to convective coefficient would be equal to the ratio of total heat-transfer surface area to inside shell area. The percentage increase in effectiveness due to the fins for ideal conduction would be about 252 percent; whereas, for the range of coolant flows considered in this analysis, the actual increase for this particular design is 225 percent at a coolant flow per blade of 0.015 pound per second and 170 percent at a coolant flow per blade of 0.08 pound per second. Thus the effectiveness of the corrugated-fin-insert design used in this analysis is high in the desired range of coolant weight flows.

Nonrefrigerated Air-Cooling-System Characteristics

The simple air-cooling system illustrated in figure 1 uses bleed air taken directly from the compressor outlet and discharged from the tip of the blade. The cooling-air temperature at the blade base was taken as 60° F higher than the coolant-supply temperature for all conditions. This value was based on experience with full-scale jet-engine installations.

Coolant-flow requirements. - The results of the analysis of the nonrefrigerated system are summarized in table III. The coolant temperature at the blade base varies from a minimum value of 465° F at a Mach number of 0.8 and an altitude of 35,000 feet to a maximum value of 862° F at 50,000 feet and a level flight Mach number of 2.5. The allowable blade temperature at one-third span, based on 100-hour stress-rupture properties of S-816 and a stress-ratio factor of 1.9, is 1210° F. The required coolant-flow ratio to maintain the desired stress-ratio factor at all flight conditions varies from 0.0182 to 0.0425, depending on the variation of outside and inside heat-transfer coefficients and the coolant temperature at the blade base with flight Mach number and altitude. As a result of the higher design blade temperature encountered in air cooling, the outside heat-transfer coefficient is less than with liquid coolants. At the combat Mach number and altitude, the outside heat-transfer coefficient is 116.6 Btu/(hr)(sq ft)(°F) as compared with 128 Btu/(hr)(sq ft)(°F) for the water-cooled turbine in reference 1. As a result of the lower heat-transfer coefficient and the smaller difference between gas and blade temperature, the ratio of heat transfer to turbine work for the air-cooled blade at combat conditions is reduced to 0.0249 as compared with 0.0554 for the water-cooled turbine (reference 1).

From table III it is evident from the required coolant-flow ratio that the critical operating condition for the nonrefrigerated air-cooling system occurs at a Mach number of 2.5 and an altitude of 50,000 feet.

The required coolant-flow ratio for the design point at combat Mach number of 1.8 and altitude of 50,000 feet and for a stress-ratio factor of 1.9 appears reasonable. The influence of the $2\frac{1}{2}$ percent compressor bleed on over-all engine performance is small for an afterburning engine. For 2758
a nonafterburning engine, the loss in thrust relative to an uncooled engine would be approximately $2\frac{1}{2}$ percent and the increase in specific fuel consumption would be negligible for the conditions of this analysis. At the maximum flight Mach number, the coolant-flow ratio of 0.0425 can probably be tolerated for most applications.

A question of interest in connection with determining minimum coolant-flow requirements, as in the analysis summarized in table III, is the influence of rotor-blade solidity and chord length on the number of blades in the rotor and the total coolant-flow requirements. Current-production engines of the approximate size considered in this analysis commonly have 100 or more rotor blades in a stage with relatively short blade chords. Because of the distribution of the local heat-transfer coefficients around the blade, the average heat-transfer coefficient of short-chord blades tends to be higher than large-chord blades. In addition, for a given size blade, the surface area and total coolant-passage area vary directly as the number of blades on the rotor. The combined effect of variation in rotor solidity and blade chord length is illustrated in figure 7, which shows clearly the reason for the large chord and relatively low solidity of the 58-blade turbine used throughout this analysis. The relative coolant-flow ratio is given as a function of the number of rotor blades with chord and solidity as parameters. If the chord length is held constant and the rotor solidity increased, the coolant-flow requirement increases very rapidly. If the solidity is held constant and the number of blades increased simply by using a short-chord blade, the coolant-flow requirement may become 40 percent greater than the basic 58-blade turbine used in the analysis. The design of cooled turbines with a large number of small blades therefore results in unnecessary penalties in coolant-flow requirements, and considerable gains can be made with designs such as the basic turbine provided that aerodynamic limitations are not exceeded with consequent loss of turbine efficiency. Little further gain could be made in the coolant-flow requirements indicated in table III by adjusting blade chord and rotor solidity because the aspect ratio of the blade is already reduced to about 2 and the number of blades reduced considerably below present practice. The aerodynamic limits of such configurations are not well established, but it is believed that for the engine design considered herein the 58-blade turbine is close to the practical limit.

The characteristics of the nonrefrigerated system given in table III represent only a reference point because additional calculations are required to evaluate the over-all suitability and probable margin of

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safety in the design. A minimum design stress-ratio factor of 1.9 at the one-third-span position was selected as a reference value on the basis of experimental investigations in a turbine of similar hub-tip ratio; it is believed to be adequate following extensive development of a blade configuration to provide uniform cooling and reliable structural features such as taper ratio, fillets, and uniform brazing and welding. In practice, the design stress-ratio factor must be evaluated from experience with each particular engine and blade configuration since the influence of gas bending loads, thermal stresses, twisting moments, and vibratory stresses vary considerably. In the present evaluation the stress-ratio factor at the one-third-span position was considered because past experience has indicated that the blade is most likely to fail at that position. It may be possible that as the general stress level of the entire blade is raised the critical section of the blade may be nearer to the root; for example, the maximum obtainable stress-ratio factor at the blade root for the blade considered in this report is 1.9. The minimum allowable stress-ratio factor in this region of the blade is unknown because experimental data are not available for air-cooled blades operating at the stress level considered herein. In addition, the stress-ratio factor is based on rupture properties of the material, which may not be significant in the failures encountered in a particular case. An example of this would be the occurrence of severe vibrations near the tip of the blade, in which case increasing the design stress-ratio factor at the one-third span would have doubtful value. An insight into the sensitivity of the particular blade design can be gained, however, by considering a range of design stress-ratio factor at the one-third-span position.

Influence of design stress-ratio factor on coolant flow. - The variation of required coolant-flow ratio with design stress-ratio factor at the one-third span is given in figure 8 for combat Mach number and altitude. It can be noted that it is practically impossible to achieve a stress-ratio factor of 2.5 by increasing the coolant-flow ratio. There are two factors which affect the maximum obtainable stress-ratio factor: (1) The stresses in the blade are such that, even if the blade metal temperature were reduced to 100° F, the stress-ratio factor at the one-third-span position would only be 2.6; (2) a point of diminishing returns is obtained with air cooling so that even very large quantities of cooling-air flow have only a small effect on reducing the blade temperature. Since the blade taper ratio is already very favorable for an air-cooled blade, there is little prospect for structural modifications to permit any increase in the maximum obtainable stress-ratio factor; however, the maximum stress-ratio factor is of little importance so long as the design stress-ratio factor is adequate for safe operation.

Decreases in the coolant flow below the design value of 0.0245 have a very marked effect on stress-ratio factor. It is therefore possible that random variations in engine operation or inadvertent over-temperature

operation could appreciably decrease the margin of safety for the blade. Consequently, it may be advisable for such designs to operate normally at a somewhat higher coolant-flow ratio as an added factor of safety.

The nonrefrigerated system of air cooling appears to be adequate for the conditions considered in this report, and the system is desirable because of its light weight and mechanical simplicity. In order to consider methods of cooling that would have a greater margin of safety, or systems that might be required for applications where stronger cooling is required, the characteristics of refrigerated air-cooling systems were considered. 2758

Refrigerated-Air-Cooling-System Characteristics

Two possibilities exist to improve the margin of safety in an air-cooled turbine: The first is substitution of an improved blade configuration, such as the strut-blade design discussed in references 15 and 16, which can tolerate higher coolant temperatures; the second is to apply bleed aftercooling or refrigeration to reduce coolant temperature, thereby either reducing coolant-flow requirements or increasing the stress-ratio factor according to the particular needs. In order to provide a basis for evaluating bleed aftercooling and refrigeration, calculations were made for the combat Mach number and altitude to determine the influence of the coolant-supply temperature on the coolant-flow ratio at a given stress-ratio factor and to determine the influence of the coolant-supply temperature on the stress-ratio factor at a given coolant-flow ratio.

Influence of coolant temperature on coolant-flow ratio and stress-ratio factor. - The variation of required coolant-flow ratio with inlet cooling-air temperature at the blade base is given in figure 9(a). For a constant design stress-ratio factor of 1.9 at combat Mach number and altitude, the required coolant-flow ratio decreases from 0.0245 at 664° F inlet coolant temperature to 0.0126 at 100° F inlet coolant temperature. The difference in engine performance between these two extremes of coolant-flow ratio would be negligible in an afterburning engine, and the gain in thrust of the lower coolant flow would only be on the order of 1 percent for a nonafterburning engine. Of greater significance is the rise in coolant-flow ratio as the air temperature exceeds the design value of 664° F for the simple air-cooling system with compressor-outlet bleed. An increase of only 100° F in compressor-outlet-bleed temperature as a result of atmospheric variations or off-design flight conditions would result in almost a 25-percent increase in coolant-flow ratio to maintain a constant stress-ratio factor of 1.9. This trend is also shown in figure 9(b), which gives the stress-ratio-factor variation with coolant inlet temperature for the same conditions as the previous figure except that the coolant flow is held constant at the design value of

0.0245 for the basic turbine. The curve shows that increases in coolant temperature can reduce the stress-ratio factor to values which may be too low for reliable operation. Figure 9(b) also indicates the influence of coolant temperature reduction on stress-ratio factor. A reduction in temperature of 200° F increases the stress-ratio factor to almost 2.2 and provides a greater margin of safety over the design value of 1.9. As shown on figure 8, an increase in coolant-flow ratio to approximately 0.035 would be required to obtain a similar stress-ratio factor with
2758 compressor-outlet-bleed air. The additional loss in thrust for a nonafterburning engine using outlet bleed with a design stress-ratio factor of 2.2, relative to the same stress-ratio factor with refrigerated cooling air, is about 1 percent for the conditions cited. Thus, bleed refrigeration offers an alternate method of increasing the design stress-ratio factor without losses associated with higher coolant-flow ratio. The main purpose of bleed refrigeration is not to reduce required coolant-flow ratio for a given minimum design stress-ratio factor, but to increase the margin of safety and provide for off-design operating conditions that might otherwise affect the reliability of the turbine.

Water-injection system. - The simplest method of bleed aftercooling is the injection of water to reduce the coolant temperature to saturation conditions immediately ahead of the point of introduction into the rotor. Since the water injected for this purpose is lost overboard in the exhaust, application of this system is dependent upon flight duration. Analysis of water injection for the flight and operating conditions given in table II was made and the results are summarized in table IV. The final cooling-air temperature varies somewhat according to the temperature and pressure at compressor outlet, but a coolant-supply temperature to the rotor of 166° F was obtained at combat Mach number and altitude. It was assumed that the coolant-flow ratio could be reduced to 0.014 at this air temperature, and the water-to-air ratio for this condition was 0.097. In order to determine the total weight of water required for the mission, the average water rates during the acceleration and climb phases of the flight were evaluated. The water rate at combat conditions was constant. As shown in table IV, the total weight of water required for the flight was 93.3 pounds per engine. This method of coolant-bleed refrigeration appears promising for missile-engine application where mechanical simplicity in both the turbine rotor and the air-supply system would be desirable.

Bleed-aftercooling system. - A number of alternate bleed refrigeration systems having equal or greater cooling effectiveness compared with that of water injection are available, but further increase in complication is encountered in making the system an integral part of the engine. The next step that can be applied in afterburning engines is to reject heat to the afterburner fuel in a bleed aftercooler. Aftercooler design conditions for a 0.5-square-foot tubular heat exchanger used to obtain the same temperature reduction as with water injection are given in

table V. The rate of heat rejection in this case is 108 Btu per second as compared with 291 Btu per second for the fuel-cooled turbine (reference 1) so that the heat-rejection rates to the fuel and also to the fuel tank for nonafterburning operation with bleed aftercooling are considerably less than they are when the entire turbine-cooling load is rejected to the afterburner fuel flow. In table V, the maximum temperature rise of the fuel occurs at the combat Mach number and altitude, which were selected as the design point for the aftercooler. At a given operating point the afterburner fuel flow and the cooling-air weight flow are fixed, and it is necessary to solve for the required core depth 2758 in the aftercooler to meet the given heat-rejection rate. The operating points of the 0.5-square-foot section are given in figure 3 for the values of table V. A 12-inch core depth was found to be inadequate and the characteristics given in the figure indicate that approximately an 18-inch core depth is required for this application. The required cooling-air temperature at the design point was 166° F, the temperature rise of the afterburner fuel was 98° F, and the estimated pressure drop through the core section was approximately 4 inches of water because of the high pressure and density with the compressor-outlet air. The main purpose of the analysis was to obtain a weight comparison between water injection and bleed aftercooling since the short duration of the mission appeared to permit application of water injection. The estimated weight of the bleed aftercooler filled with fuel was 78 pounds, indicating a considerable difference in weight as compared with the water injection system which required 93 pounds of water plus the tank weight.

Comparison of the two flight speeds considered at an altitude of 35,000 feet in table V indicates that performance of the bleed aftercooler becomes more critical as flight Mach number is increased because of the higher coolant temperature reduction desired. With a given core size, the coolant-supply temperature to the turbine would increase eventually as flight Mach number increased in spite of the stable temperature of the heat-rejection medium. The extent to which engine operation would be limited by this effect at high flight Mach number was not determined although a considerable margin appears available in the 0.5-square-foot section used in table V.

Air-bleed expansion refrigeration systems. - In all cases considered in this analysis, the cooling air is taken at compressor-outlet conditions and, since the supply pressure required at the point of introduction of the cooling air into the turbine rotor is lower than the compressor-outlet pressure, the possibility exists of operating an expansion refrigeration cycle between these points. A distinction is drawn between two modifications of the refrigeration cycle depending upon how the expansion turbine is loaded. In the open cycle, illustrated in figure 2(b), the turbine is loaded by an aircraft or engine accessory such as the afterburner fuel pump, and the work of expansion leaves the system. In the closed cycle, illustrated in figure 2(c), the expansion turbine is loaded

by an auxiliary compressor that is in series with the aftercooler and expansion turbine, and the work of expansion is ultimately rejected as heat in the bleed aftercooler. The thermodynamic cycle of the open and closed systems is indicated in figure 10. The characteristics of the two systems at combat Mach number and altitude are given in figure 11, in which cooling-air-supply temperature is plotted against aftercooler-outlet temperature. The 45° line indicates the cooling-air-supply temperature without use of an expansion turbine; that is, air is taken directly from the compressor outlet or it is cooled below the compressor-outlet temperature in an aftercooler. The curve labeled "Expansion through auxiliary turbine" gives the possible operating points of the open system in which a fixed pressure ratio is available for the expansion turbine at any given operating condition. For the basic engine, the cooling-air-supply temperature can be varied from 440° F with no aftercooling to 50° F for the 166° F aftercooler-outlet temperature given in the preceding example. The temperature reduction of the open system is limited by the pressure ratio available so that loading of the turbine becomes a problem because of the large variation in output as flight conditions vary. Perhaps the only suitable auxiliary would be the afterburner fuel pump because its power requirements vary in a similar manner to the turbine-cooling requirements. The results in figure 11 show the manner in which expansion refrigeration extends the Mach number capability of an air-cooling system. Even with no aftercooling at combat Mach number and altitude, a reduction in cooling-air-supply temperature to 440° F is possible and, if the compressor-outlet temperature increased to 800° F at higher Mach number, the coolant-supply temperature could be reduced to 600° F with no aftercooling, in which case the margin of safety of the turbine would be no less than the nonrefrigerated air-cooling system at the lower design-point flight Mach number.

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The lower curve in figure 11 gives the performance of the closed cycle under the same conditions. The advantage of the closed cycle is small at very low aftercooler-outlet temperatures, but it can be seen that the amount of aftercooling required to achieve a given cooling-air-supply temperature is considerably less with the closed cycle over most of the range considered. This circumstance, in conjunction with the higher pressures following precompression, would probably result in a smaller aftercooler and thereby offset part of the weight of the auxiliary compressor. The outstanding difference between open and closed cycles occurs at high aftercooler-outlet temperatures, for example, at 500° F where an additional temperature reduction of 140° F is possible.

Regenerative air-bleed expansion refrigeration system. - The regenerative bleed expansion refrigeration system evolves from the closed cycle of the preceding discussion and is represented by that portion of the lower curve in figure 11 to the right of a 604° F aftercooler-outlet temperature. The thermodynamic cycle of the regenerative system is indicated in figure 10(c). The characteristic of the system is such that,

if the aftercooler is relocated in the main-engine working fluid at compressor-outlet conditions and the cooling air reduced to within 50° F of engine compressor-outlet temperature following precompression in the auxiliary compressor, the final coolant-supply temperature to the turbine rotor can be reduced by 314° F to a value of 290° F. Thus, the expansion system can be made completely independent of an auxiliary system such as the afterburner fuel system and is suitable for nonafterburning engines. The flight Mach number capability of the regenerative system appears good, and the curve in figure 11 indicates that at a compressor-outlet temperature of 802° F at the maximum level flight Mach number of 2.5 a 2758
turbine-coolant-supply temperature of 354° F can be maintained. Thus, with the regenerative system, a given margin of safety can be retained in spite of increased flight Mach number and other off-design operating conditions. A weight estimate for the components of the expansion refrigeration systems was not determined, but similar equipment is currently under intensive development for aircraft applications.

COMPARISON OF TURBINE AIR- AND LIQUID-COOLING SYSTEMS

An analysis of turbine liquid-cooling systems is presented in reference 1 for identical turbine design and operating conditions and is used as a basis for comparison with the air-cooling systems considered in this report. The combined results of the two reports therefore permit direct comparison of air and liquid systems on the basis of the most desirable characteristics for a given heat-rejection medium, either ram air, afterburner fuel, or main-engine working fluid.

The simple nonrefrigerated air-cooling system considered in this report was found to be adequate for all conditions, although the coolant-flow-ratio requirements at supersonic speed almost doubled when the flight Mach number at 50,000 feet was increased from 1.8 to 2.5. This system appears to represent the least mechanical complication in installation and is independent of heat-rejection problems encountered in other more complex systems. A limitation that is encountered, however, is that the simple system is more sensitive to variations in coolant-supply temperature at high flight Mach number and generally has a lower margin of safety than can be provided in other systems. Since compressor-bleed air is inherently less effective than liquids for turbine-blade cooling, the only methods available for improving the basically simple air-cooling system are either to greatly increase the heat-transfer effectiveness within the coolant passage by means of extended surface or other devices, or to provide special methods for reducing the cooling-air-supply temperature to the blade. Bleed aftercooling with ram air is a possible method, but introduces almost the same external complication in ducting and auxiliary apparatus as the liquid-cooled turbine with ram radiator.

The water-cooling system with heat rejection to ram air in a radiator duct, investigated in reference 1, is limited by a considerable weight

penalty and the possibility of severe drag losses at off-design radiator operating conditions, although a satisfactory radiator design appears practical at combat conditions. The ability of the water-cooling system to accommodate the maximum level flight Mach number of the aircraft was inadequate because of heat-rejection consideration in the radiator at high ram temperature and resultant excessive pressures in the system. The over-all suitability of the water-cooling system was further limited by vulnerability, the undesirable influence on aircraft configuration resulting from large radiator frontal area, and installation complexity of the radiator, pumps, plumbing, and radiator ducting which are external to the main engine and therefore influence the aircraft design to a considerable extent.

For afterburning engines, an alternate possibility is the fuel-cooling system considered in reference 1, although a serious limitation exists if provision must be made for periods of nonafterburning operation during cruising portions of the flight. The design, installation, and mechanical problems of the fuel-cooling system are relatively simple, and weight and performance penalties appear to be negligible. The temperature of the heat-rejection medium in this case is stable and the system is therefore insensitive to high flight Mach number. The fuel-cooling system can be applied in the supersonic interceptor with limited flexibility of operation, although provision must be made for pressurization of the fuel tanks to avoid evaporation losses during portions of the flight where heat is rejected temporarily to the tanks. The system is therefore limited in interceptor-aircraft application because of the undesirable influence of tank pressurization on aircraft configuration, fuel-tank design and installation problems, and increased vulnerability of the pressurized fuel tanks. An additional uncertain limitation is the chemical stability of the fuel under the conditions imposed in a turbine-cooling system, but the fuel-cooling system otherwise appears desirable and applicable for supersonic speed and high-altitude flight.

Bleed refrigeration systems that permit reduced coolant flow or increased margin of safety of the air-cooling system at high flight Mach number were considered. These systems used afterburner fuel as the heat-rejection medium and were therefore subject to many of the installation and operating complexities of the fuel-cooled turbine if operation with and without afterburning was required in the flight plan. Application of aftercoolers using fuel as the heat-rejection medium to the air-cooling system is a desirable alternate for the direct-fuel-cooling system. Of the systems investigated in this report and reference 1, all the air-cooling systems and the direct-fuel-cooling system can accommodate the maximum level flight Mach number, but the air-cooling systems require the least additional development and have a minimum of new fabrication problems in any given engine since air cooling permits extension of a fully developed turbine-rotor design to more severe operating conditions with revision only of the auxiliary components of the engine. The most

promising application for turbine-cooling systems based on heat rejection to the afterburner fuel is the guided-missile engine at very high flight Mach number and for missions in which the afterburner operates continuously throughout the flight, thus avoiding heat rejection to the fuel tanks.

The regenerative liquid-cooling system appears promising for eventual application in the supersonic interceptor aircraft and can reject 2758
heat to the main-engine working fluid at compressor outlet up to the maximum level flight Mach number. The principal problem in its application is selection of a suitable fluid to operate within the desired temperature and pressure limits and, under some conditions, a vapor refrigeration cycle operating off the main-engine turbine may be required.

The regenerative air-cooling system is based on well-developed concepts in expansion refrigeration cycles except that heat rejection occurs at the compressor outlet in the same manner as the liquid system. The air-cooling system appears basically simpler than the liquid system for regenerative cooling, and both turbine-rotor weight and the weight of auxiliary parts of the system would probably be less. The coolant refrigeration capacity of the expansion system provides an adequate margin of safety for an air-cooled turbine up to a maximum level flight Mach number of 2.5. Although both air and liquid regenerative cooling systems are suitable for the supersonic interceptor aircraft, it is believed that the air-cooling system provides the least mechanical complication and operating problems.

It was found that acceptable flight performance of the supersonic interceptor aircraft assumed in this analysis could be obtained with improved engine designs having increased specific mass flow, turbine-inlet temperature, and blade tip speed. The basic engine design specifications selected for the analysis resulted in turbine stresses and operating conditions that were beyond the capacity of the best uncooled high-temperature-alloy turbine-blade materials. Thus applicability of improved turbojet engines at supersonic flight speed depends to a considerable extent upon improved turbine-cooling systems such as those considered in this analysis. The comparison of air- and liquid-cooling systems showed that adequate cooling effectiveness could be achieved in spite of the high ram and compressor-discharge temperatures encountered at supersonic speed, and the ultimate limits of air and liquid systems were not encountered within the specified flight conditions of Mach number 2.5 and altitude of 50,000 feet.

In conclusion, when the over-all picture of research and application is considered, the analysis indicates that, for the present, intensive development of the simple air-cooling system should be continued because it is capable of operation at supersonic speed and high altitude, and because such basic developments are directly applicable in improved

systems with bleed aftercooling. Since turbine-inlet temperature can not be raised sufficiently for some time to avoid necessity for afterburning in interceptors and guided missiles, development would proceed in the direction of bleed aftercooling systems and bleed-expansion refrigeration systems based on heat rejection to the afterburner fuel. In this manner a fully developed air-cooled-turbine configuration suitable for less-severe operating conditions could be extended to higher flight Mach number without basic changes. Also, for afterburning engines with moderate increase in turbine-inlet temperature, the fuel-cooling system using afterburner fuel is a promising development, particularly for very high flight Mach number. As soon as parallel development of the stationary parts of the turbojet engine permits operation at sufficiently high gas temperature so that afterburning can be avoided, application of the regenerative air- and liquid-cooling systems may be required because of insufficient fuel flows for adequate bleed aftercooling or direct liquid cooling. Promising systems for long-range future development and application therefore appear to be the regenerative air and liquid systems which avoid any influence on other aircraft installations and systems, are applicable with nonafterburning engines as well as afterburning engines, and appear capable of operation over the desired range of flight conditions without limitations on cruising endurance or other aircraft operational conditions.

RESULTS AND CONCLUSIONS

The results of an investigation conducted to evaluate the effects of high-altitude supersonic flight on the effectiveness of turbine air- and liquid-cooling systems are as follows:

1. The coolant-flow requirements for turbine-rotor blades can vary as much as 40 percent by variations in the number and size of turbine blades that are designed for a specified work output. A small number of large blades required less coolant that a large number of small blades at the same solidity.

2. A simple nonrefrigerated air-cooling system was found to be adequate for turbine-rotor blade cooling at all conditions considered, although the required coolant-flow ratio doubled from flight Mach number of 1.8 to flight Mach number of 2.5. Further development of the simple air-cooling system is desirable and also directly applicable to improved systems with bleed aftercooling or refrigeration.

3. Air-cooling systems with bleed aftercooling or refrigeration based on heat rejection to afterburner fuel provide a greater margin of safety than the simple nonrefrigerated systems, extend operation to higher permissible flight Mach number without basic change in the engine, and provide the least mechanical complication and operating problems.

4. The most desirable alternate choice for supersonic interceptor and guided-missile engines is the liquid-cooling system based on heat rejection to the afterburner fuel, particularly for very high flight Mach number provided that limitations can be placed on the required cruising endurance. Of the liquid-cooling systems considered, this system has the least weight and performance penalties and is relatively simple to install. 2758

5. The most promising applications for turbine air- and liquid-cooling systems based on heat rejection to the afterburner fuel are for missions in which the afterburner operates continuously throughout the flight, thus avoiding heat rejection to the main fuel tanks.

6. Regenerative air- and liquid-cooling systems rejecting heat to compressor-outlet air can be provided for nonafterburning engines where fuel flows may be insufficient for adequate bleed refrigeration or direct-fuel cooling.

7. The regenerative air-bleed expansion refrigeration system is a promising long-range development capable of operation over the desired range of flight conditions without influence on other aircraft installations and systems.

8. Applicability of turbojet engines at supersonic flight speed depends to a considerable extent upon improved turbine-cooling systems to permit increased turbine-inlet temperature, specific mass flow, and blade tip speed; and the analytical comparison of applicable systems showed that the ultimate limits of air- and liquid-cooling systems were not encountered for flight Mach number to 2.5 at altitudes to 50,000 feet.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio

APPENDIX - SYMBOLS

The following symbols were used in this report:

A	cross-sectional area through blade coolant passage, sq ft
b	blade height or span, ft
c_p	specific heat of gas at constant pressure, Btu/(lb)(°F)
c_v	specific heat of gas at constant volume, Btu/(lb)(°F)
D	hydraulic diameter of blade coolant passage, ft
g	acceleration due to gravity, 32.17 ft/sec^2
H	average heat-transfer coefficient, Btu/(sec)(sq ft)(°F) unless otherwise specified
H_f	average effective inside heat-transfer coefficient, Btu/(sec)(sq ft)(°F) unless otherwise specified
h'	absolute stagnation enthalpy, Btu/lb
k	thermal conductivity, Btu/(sec)(sq ft)(°F/ft)
l	blade-shell perimeter, ft
p	absolute static pressure, lb/sq ft
p'	absolute stagnation pressure, lb/sq ft
Re	Reynolds number
T	static temperature, °R or °F
T'	stagnation temperature, °R or °F
w	weight flow of fluid, lb/sec
x	distance from blade base, ft
γ	ratio of specific heats, c_p/c_v
Δ	difference
η	efficiency

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λ $H_o l_o/H_f l_i$

μ absolute viscosity of fluid, (lb)(sec)/sq ft

ρ density of fluid, slugs/cu ft

σ ρ/ρ_0

φ temperature difference ratio, $(T_{g,e} - T_B)/(T_{g,e} - T_{a,e,h})$ 2758

Subscripts:

a cooling air

B blade (when used with fluid properties or in heat-transfer equations, subscript B indicates "based on average blade temperature at critical section")

c auxiliary compressor

e effective

g combustion gas

h blade base

i inside blade surface

o outside blade surface

t auxiliary turbine

0 NACA sea-level air

1 engine compressor inlet

2 engine compressor outlet

3 auxiliary compressor inlet

4 auxiliary compressor outlet

5 aftercooler inlet

6 aftercooler outlet

7 auxiliary turbine inlet

8 auxiliary turbine outlet

9 turbine inlet

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TABLE I - SUMMARY OF CONDITIONS AND ASSUMPTIONS

Flight conditions	
Mach number range	0 - 2.5
Altitude range, ft	0 - 50,000
Combat conditions	
Altitude, ft	50,000
Mach number	1.8
Maneuverability, g	2
Loiter conditions	
Altitude, ft	35,000
Mach number	0.8
Basic engine specification	
Sea-level specific mass flow, lb/(sec)(sq ft)	23.6
Turbine-inlet temperature, °F	2040
Effective gas temperature, °F	1760
Sea-level compressor pressure ratio	6
Afterburning temperature, °F	3040
Aircraft specifications	
Gross weight, lb	28,000
Structure-to-gross-weight ratio	0.3
Pay load, lb	3,000
Unaugmented specific engine weight, lb/lb thrust	0.28
Turbine specifications	
Tip speed, ft/sec	1,500
Tip diameter, in.	35.1
Hub-tip ratio	0.732
Blade material	S-816
Stress-ratio factor	1.9
Blade critical section	1/3 span
Blade cross-section-area ratio, tip to root	0.5
Blade chord, in.	2.27
Blade solidity	1.366
Blade span, in.	4.72
Number of blades	58
Assumptions in air-cooling systems	
Temperature rise of cooling air from rotor hub to blade base, °F	60
Aftercooler fuel-inlet temperature for comparison with water-injection cooling, °F	50
Auxiliary compressor and turbine efficiency used in expansion refrigeration system	0.85

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TABLE II - ENGINE OPERATING CONDITIONS

Altitude, ft	0		35,000		50,000	
Flight Mach number	0	0.8	0.8	1.8	1.8	2.5
Compressor pressure ratio	6.0	5.2	7.0	4.6	4.6	2.6
Compressor efficiency	0.805	0.821	0.780	0.831	0.831	0.704
Compressor-outlet temperature, °F	484	547	405	604	604	802
Compressor weight flow, lb/sec	158.0	199.3	58.6	189.7	72.4	82.7
Turbine-inlet temperature, °F	2040	2040	2040	2040	2040	2040
Afterburner temperature, °F	3040	3040	3040	3040	3040	3040
Ram temperature, °F	59	126	-17	186	186	423
Primary-burner fuel flow, lb/sec	4.06	4.92	1.58	4.53	1.73	1.73
Afterburner fuel flow, lb/sec	4.88	6.04	1.80	5.50	2.21	2.46
Turbine efficiency	.83	.83	.83	.83	.83	.83
Turbine pressure ratio	2.14	2.17	2.14	2.14	2.14	2.05
Specific fuel consumption, lb/lb-hr	2.11	2.48	2.14	2.09	2.14	2.40
Thrust, lb	15,283	15,950	5665	17,706	6793	6289

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TABLE III - HEAT-TRANSFER CHARACTERISTICS OF SYSTEMS USING NONREFRIGERATED COOLING AIR BLED FROM COMPRESSOR OUTLET

[Turbine-inlet temperature, 2040° F; stress-ratio factor, 1.9; blade material, S-816; allowable blade metal temperature at one-third span, 1210° F.]

Altitude, ft	0		35,000		50,000	
Flight Mach number	0	0.8	0.8	1.8	1.8	2.5
Cooling-air temperature at blade base, °F	544	607	465	664	664	862
Average outside heat-transfer coefficient, Btu/(hr)(sq ft)(°F)	195.8	228.1	101.0	232.6	116.6	127.5
Required coolant-flow ratio	0.0182	0.0208	0.0190	0.0238	0.0245	0.0425
Heat-transfer rate to cooling air, Btu/sec	309.5	361.0	160.0	367.5	184.5	201.5
Ratio of heat transfer to turbine work q/Ω	0.0187	0.0176	0.0268	0.0189	0.0249	0.0256

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TABLE IV - HEAT-TRANSFER CHARACTERISTICS OF A SYSTEM USING VAPORIZING WATER

SPRAY TO REFRIGERATE COOLING AIR

Altitude (ft)	Flight Mach number	Compressor-outlet temperature (°F)	Compressor-outlet pressure (lb/sq in.)	Final cooling-air saturation temperature (°F)	Water-to-air ratio	Cooling-air weight flow for a coolant-flow ratio of 0.014	Duration of each phase of flight plan (sec)	Weight of water required (lb)
0	0	484	88.2	184	0.066	2.21	10.5	1.53
Accelerate to flight Mach number of 0.8							22.6	3.95
0	0.8	547	111.1	202	0.074	2.78		
Climb to 35,000 ft at flight Mach number of 0.8							75.9	9.00
35,000	0.8	405	35.1	141	0.058	0.82		
Accelerate to flight Mach number of 1.8 at 35,000 ft							85.3	11.00
35,000	1.8	604	82.6	195	0.090	2.66		
Climb to 50,000 ft at flight Mach number of 1.8							22.9	3.92
50,000	1.8	604	40.4	166	0.097	1.01		
Available combat time at 50,000 ft at flight Mach number of 1.8							651.0	63.90
Total weight of water used per engine								93.30

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TABLE V - BLEED AFTERCOOLER DESIGN CONDITIONS FOR HEAT REJECTION

TO AFTERBURNER FUEL

[Core frontal area, 0.5 sq ft; afterburner fuel-inlet temperature, 50° F; coolant-flow ratio, 0.014.]

Altitude, ft	0		35,000		50,000
Flight Mach number	0	0.8	0.8	1.8	1.8
Afterburner fuel flow, lb/sec	4.88	6.04	1.80	5.50	2.21
Engine compressor weight flow, lb/sec	158.0	199.3	58.6	189.7	72.4
Cooling-air weight flow, lb/sec	2.21	2.78	0.82	2.66	1.01
Compressor-outlet temperature, °F	484	547	405	604	604
Desired cooling-air temperature reduction, °F	300	345	264	409	438
Final cooling-air temperature, °F	184	202	141	195	166
Rate of heat exchange, Btu/sec	161	235	52.6	266	108
Temperature rise of afterburner fuel, °F	66	78	58	97	98
Heat-transfer rate per unit frontal core area of heat exchanger, Btu/(sec)(sq ft) per 100° F ΔT	80	103	32	106	43
Cooling-air weight flow per unit frontal area of heat exchanger, lb/(sec)(sq ft)	4.42	5.56	1.64	5.32	2.03

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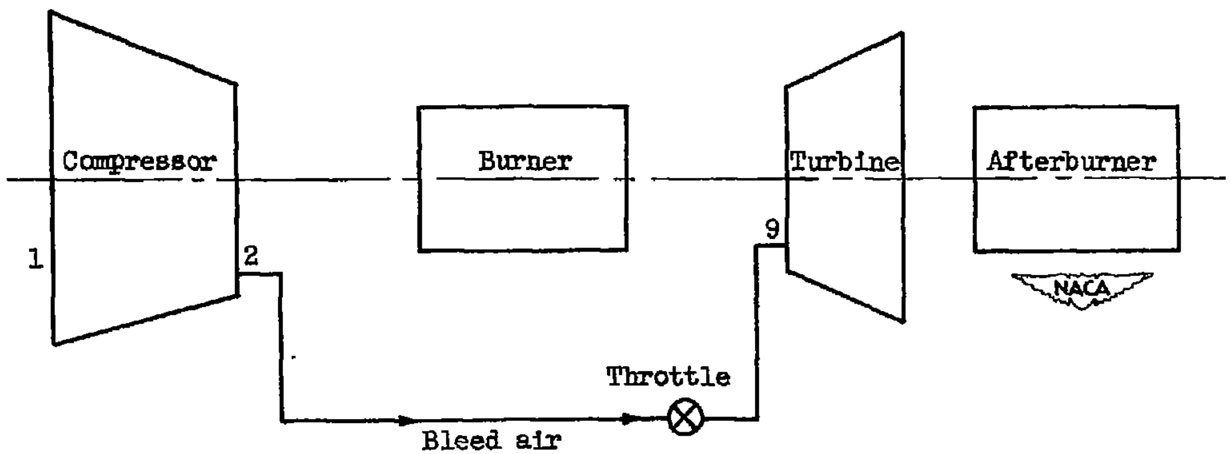

Figure 1. - Schematic diagram of simple air-cooling system.

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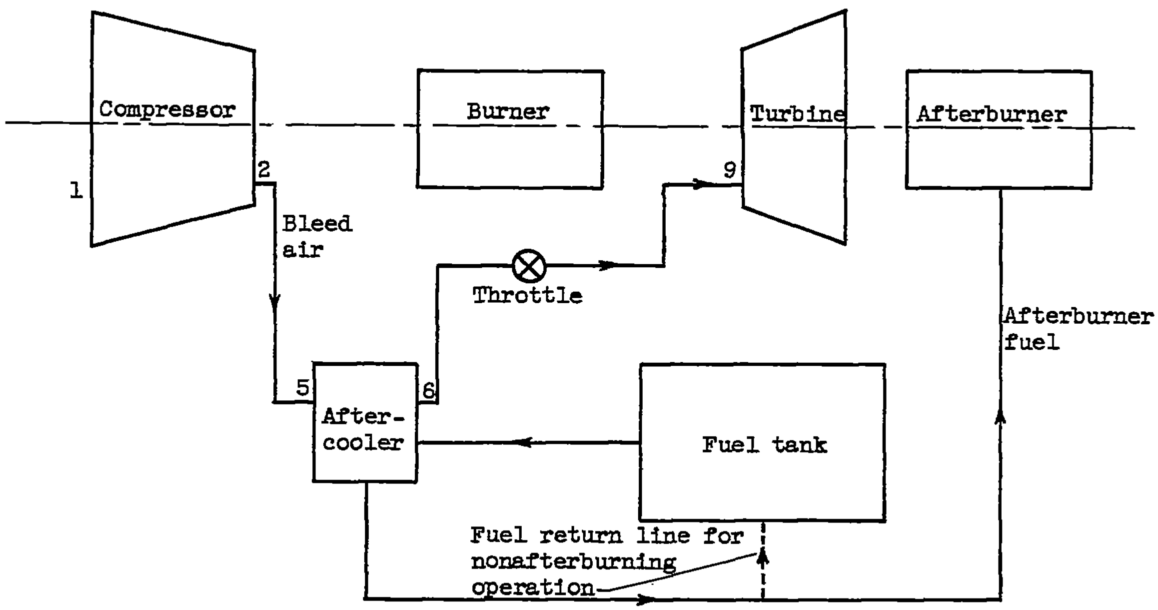
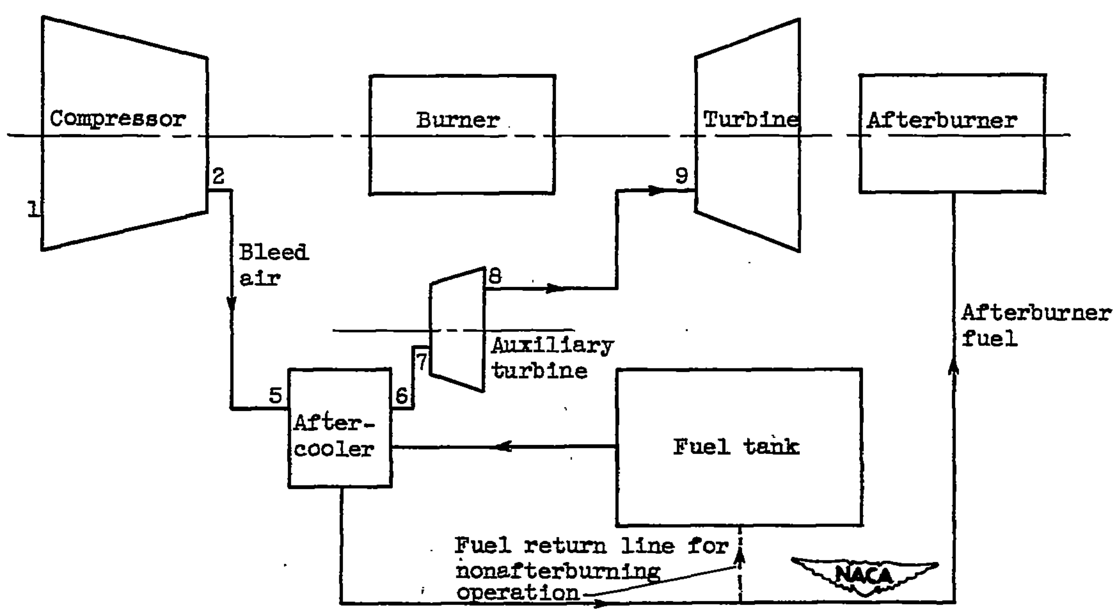
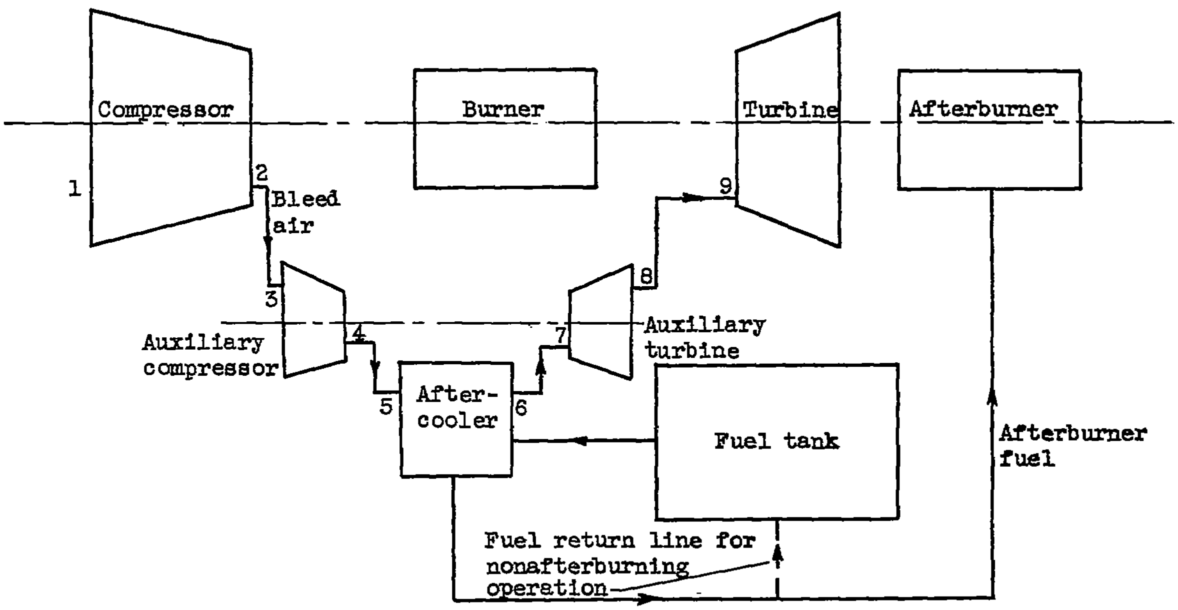


(a) Aftercooler heat exchange to fuel.

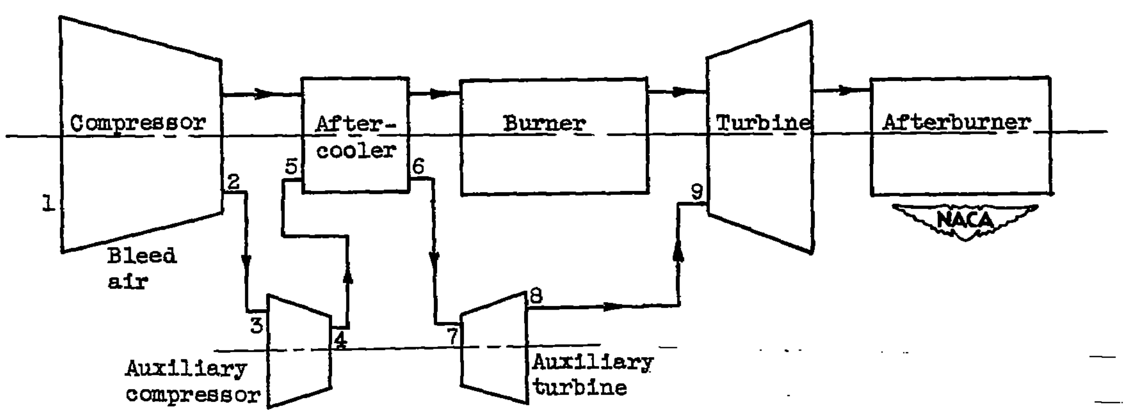


(b) Aftercooler heat exchange to fuel and expansion through auxiliary turbine.

Figure 2. - Schematic diagrams of systems used to refrigerate turbine-cooling air bled from compressor.



(c) Auxiliary compression, aftercooler heat exchange to afterburner fuel, and expansion through auxiliary turbine driving auxiliary compressor.



(d) Regenerative air-cooling system with auxiliary compression, after-cooler heat exchange to engine compressor-outlet air, and expansion through auxiliary turbine driving auxiliary compressor.

Figure 2. - Concluded. Schematic diagrams of systems used to refrigerate turbine-cooling air bled from compressor.

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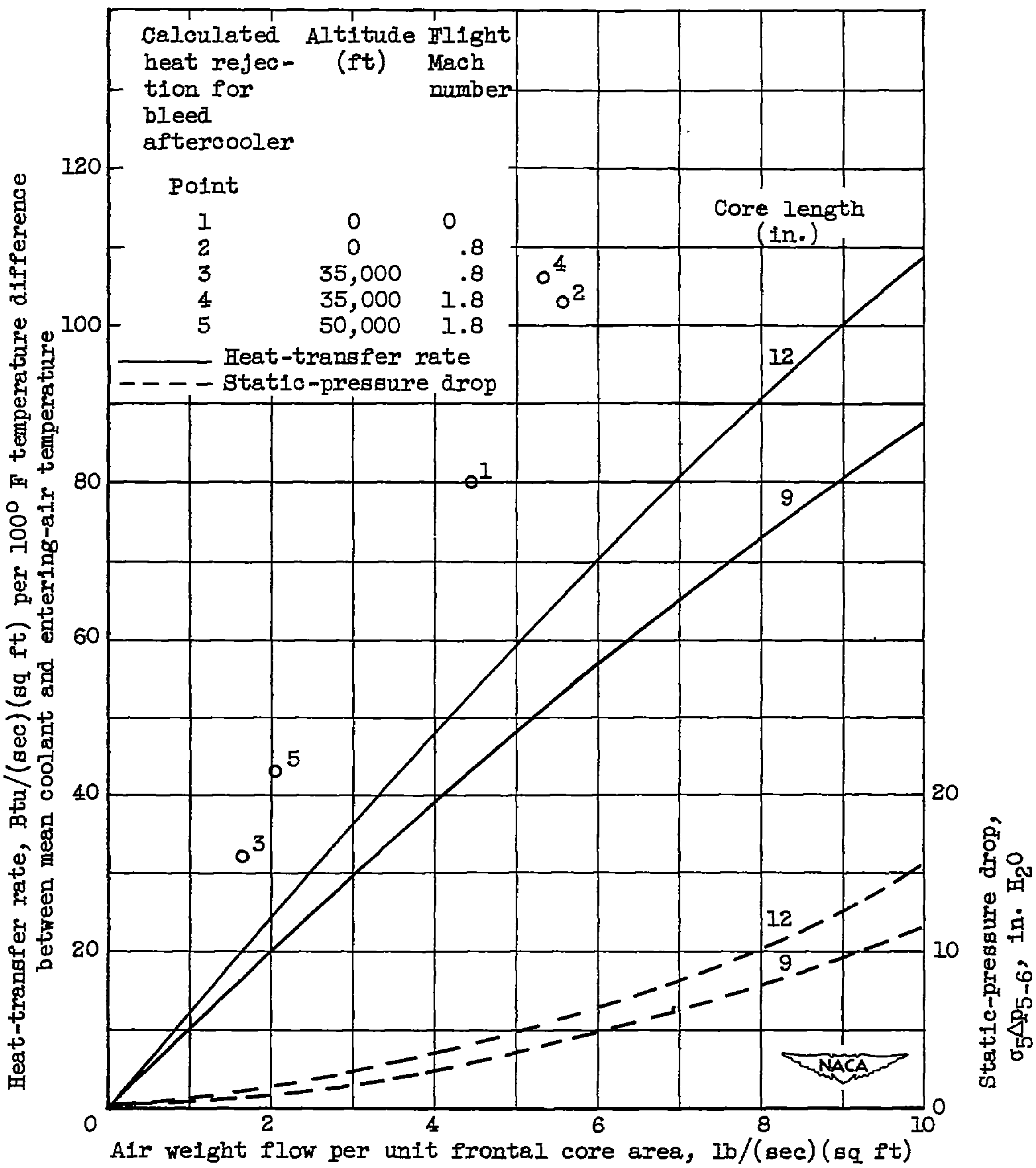

Figure 3. - Performance characteristics for 9- and 12-inch core lengths of ethylene glycol radiators and calculated heat-rejection rates for the aircraft aftercooler. (Abscissa for points is based on assumed aftercooler frontal area of 0.5 sq ft.)

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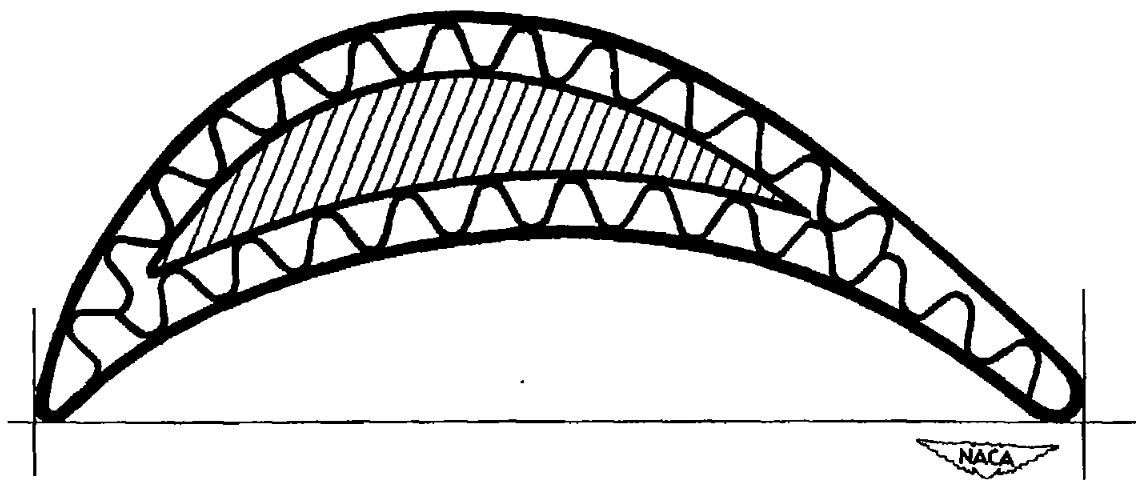

Figure 4. - Cross-sectional view of air-cooled blade assumed in analysis.

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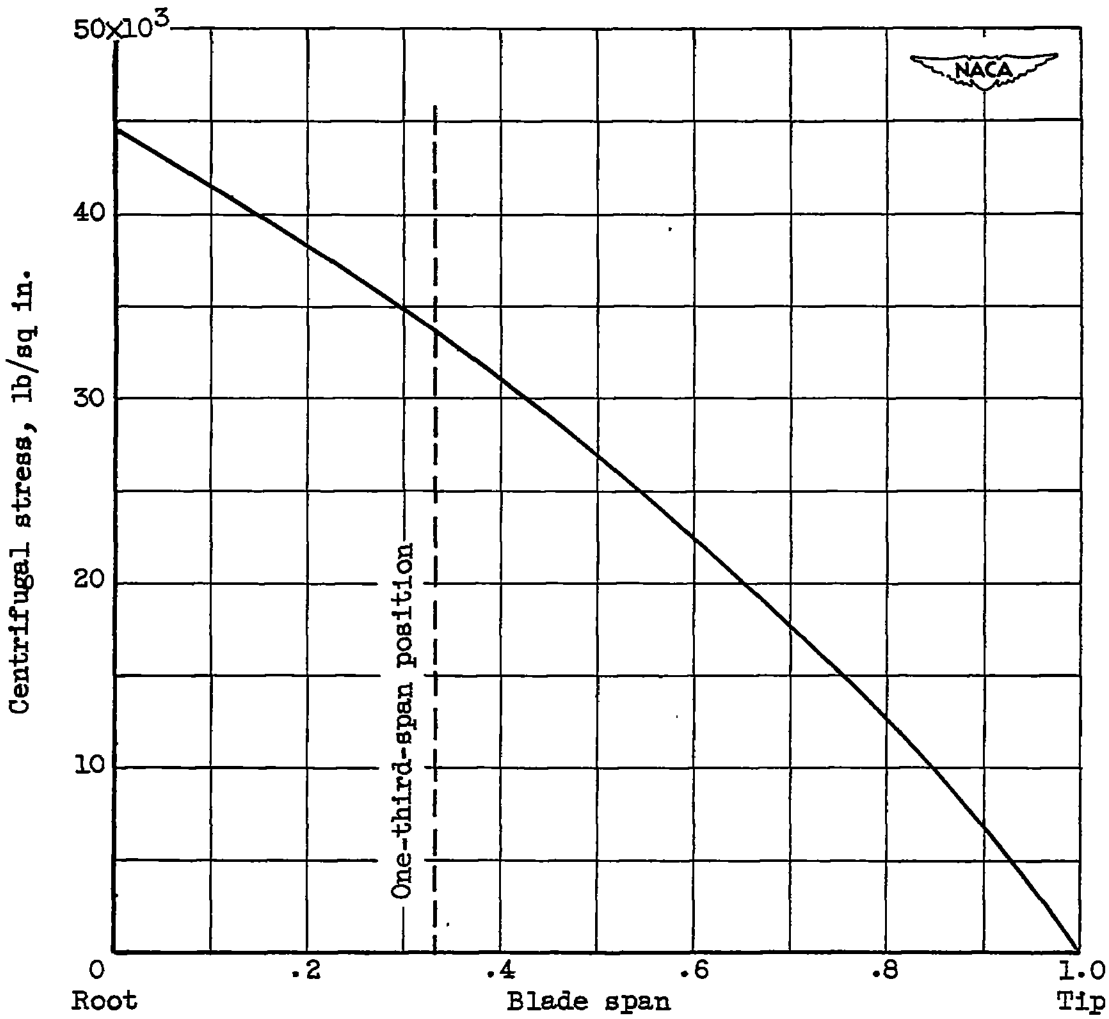

Figure 5. - Spanwise distribution of turbine-blade centrifugal stress for 0.5 ratio of tip cross-sectional area to root cross-sectional area. Blade material, S-816; hub-tip ratio, 0.732; blade span, 4.72 inches; tip speed, 1500 feet per second.

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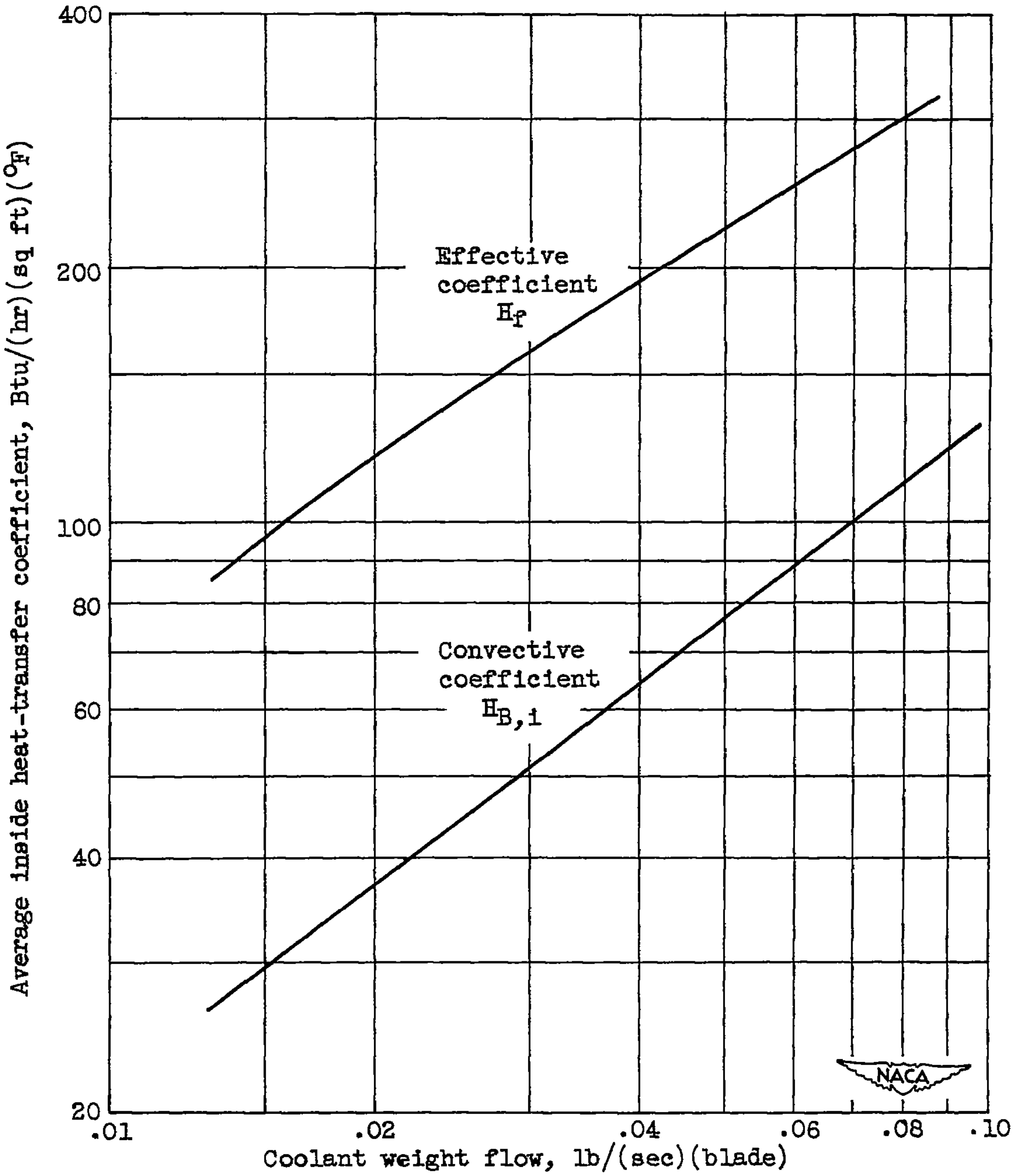

Figure 6. - Comparison of calculated effective heat-transfer coefficient with average convective heat-transfer coefficient in air-cooled corrugated blade. Average blade temperature, 1210° F; average coolant temperature, 800° F.

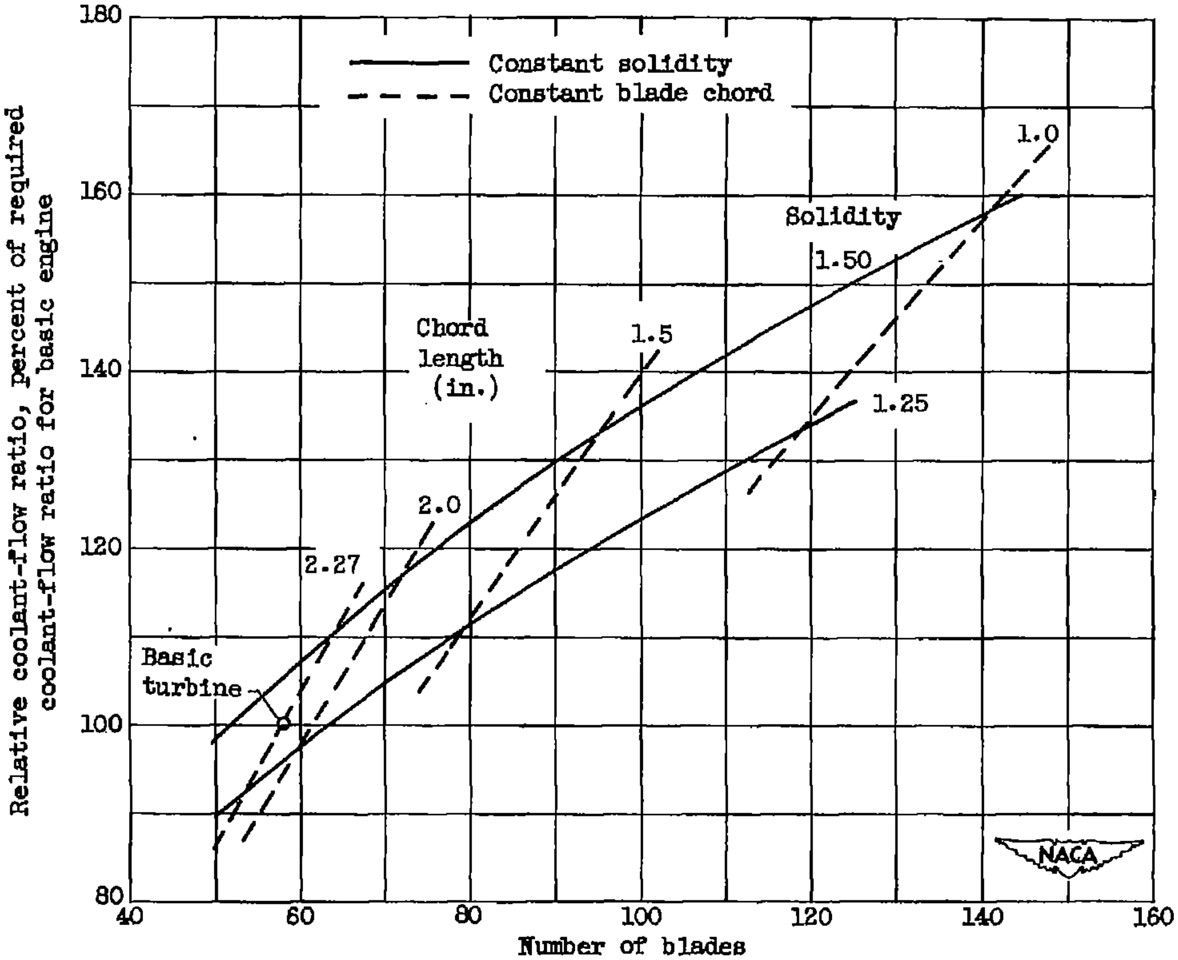

Figure 7. - Variation of relative required coolant-flow ratio with number of air-cooled blades for two values of solidity. Altitude, 50,000 feet; flight Mach number, 1.8; stress-ratio factor, 1.90; turbine-inlet temperature, 2040° F; blade material, S-816; cooling-air inlet temperature at blade base, 664° F; hub-tip ratio, 0.732; tip speed, 1500 feet per second.

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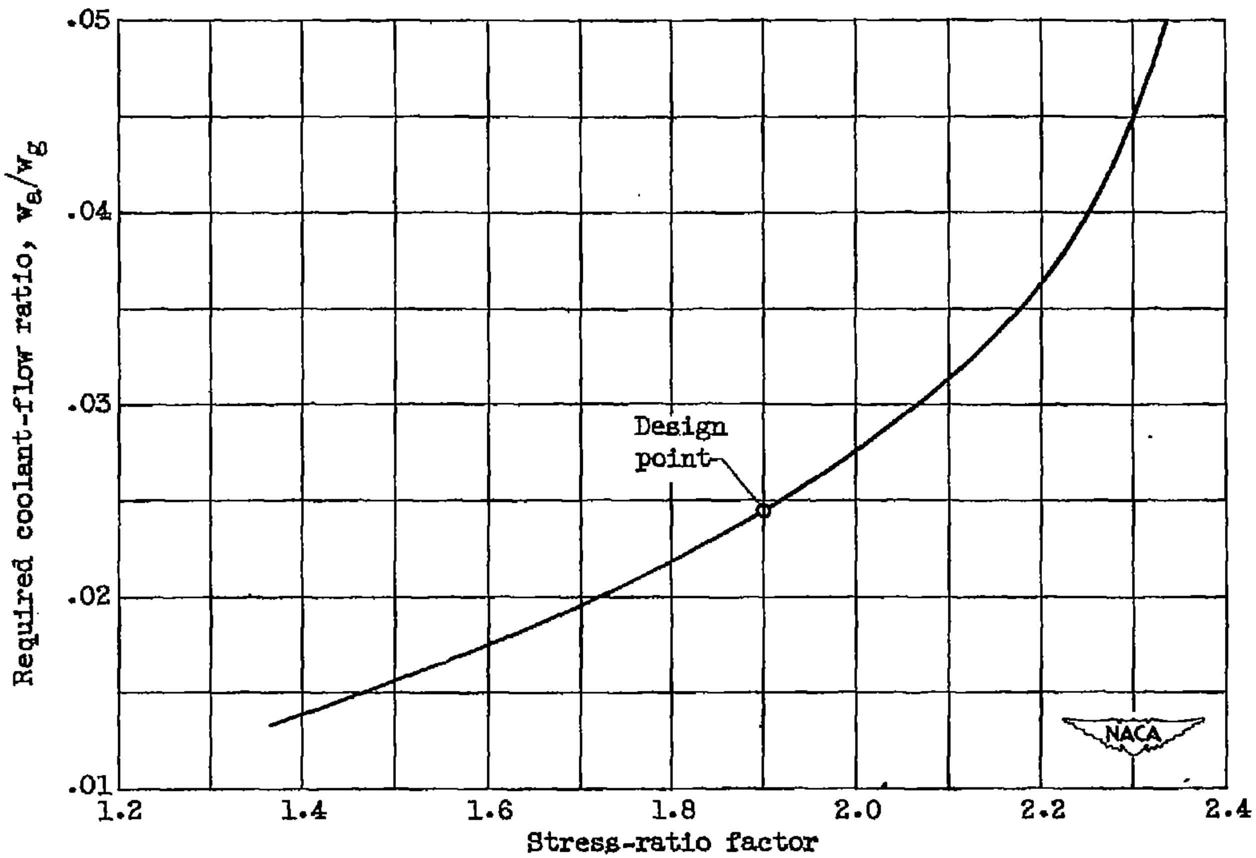
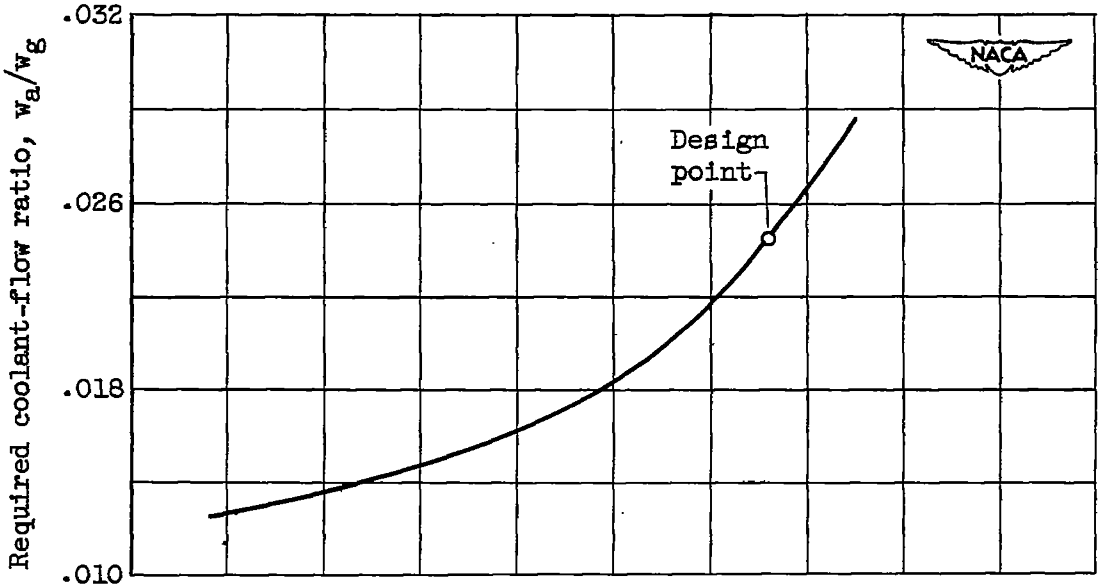
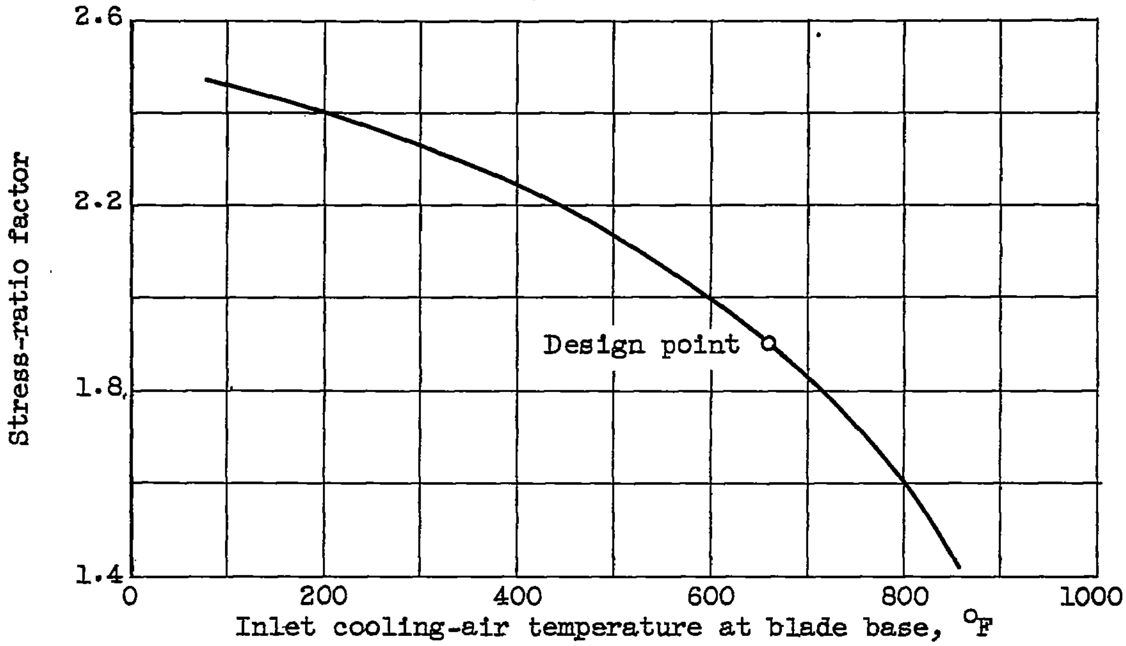


Figure 8. - Variation of required coolant-flow ratio with stress-ratio factor for air-cooled blade. Altitude, 50,000 feet; flight Mach number, 1.8; turbine-inlet temperature, 2040° F; blade material, S-816; cooling-air inlet temperature at blade base, 664° F; solidity, 1.366; number of blades, 58; hub-tip ratio, 0.732; tip speed, 1500 feet per second.



(a) Required coolant-flow ratio at a stress-ratio factor of 1.9.



(b) Stress-ratio factor at a coolant-flow ratio of 0.0245.

Figure 9. - Variation of required coolant-flow ratio and stress-ratio factor with inlet cooling-air temperature for the air-cooled blade. Altitude, 50,000 feet; flight Mach number, 1.8; turbine-inlet temperature, 2040° F; blade material, S-816; solidity, 1.366; number of blades, 58; hub-tip ratio, 0.732; tip speed, 1500 feet per second.

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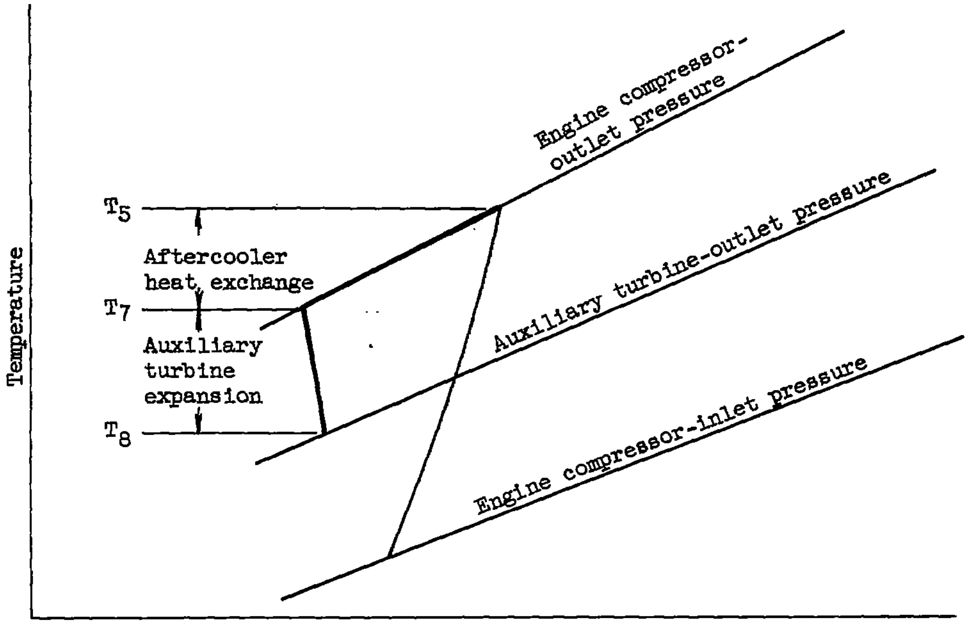
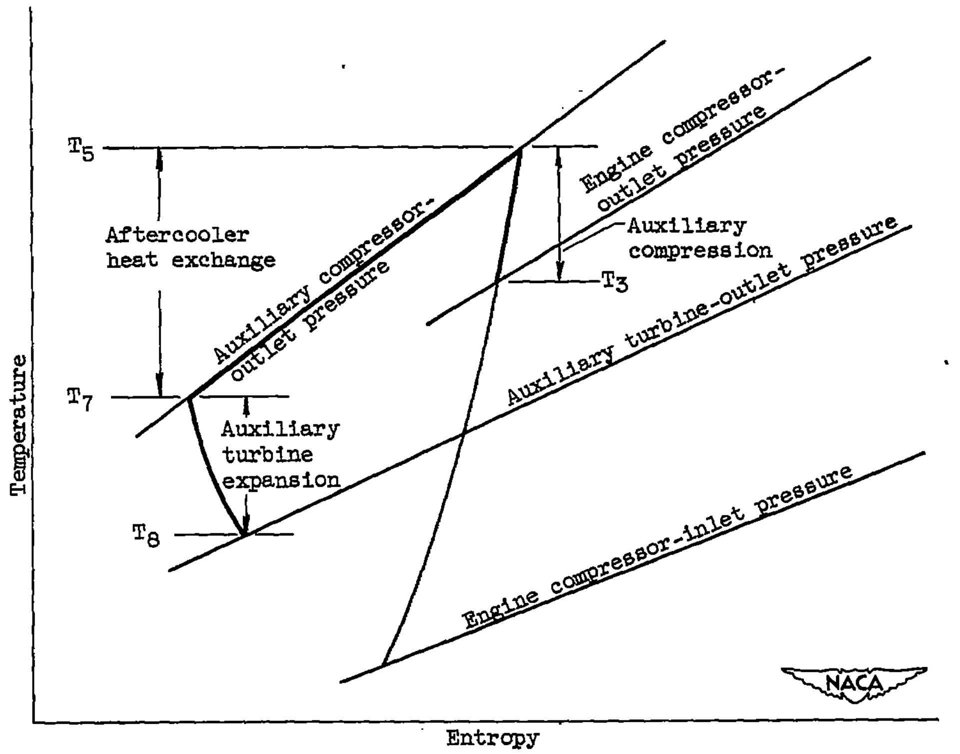
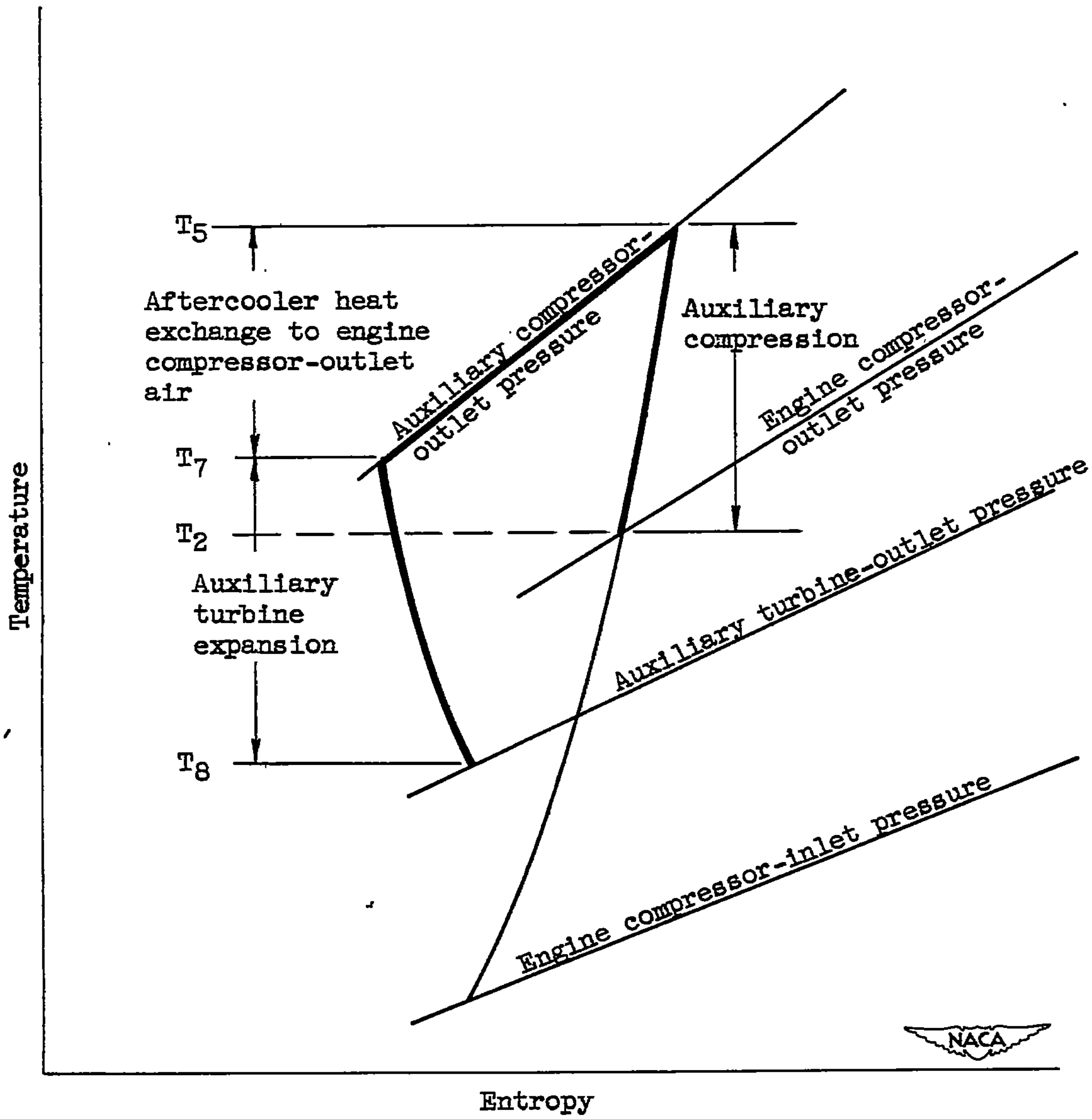


(a) Aftercooling and expansion in auxiliary turbine.



(b) Auxiliary compression, aftercooling, and expansion in auxiliary turbine.

Figure 10. - Temperature-entropy diagrams showing thermodynamic cycle of systems for refrigerating cooling air.



(c) Auxiliary compression, aftercooling by heat rejection at compressor outlet, and expansion in auxiliary turbine.

Figure 10. - Concluded. Temperature-entropy diagrams showing thermodynamic cycle of systems for refrigerating cooling air.

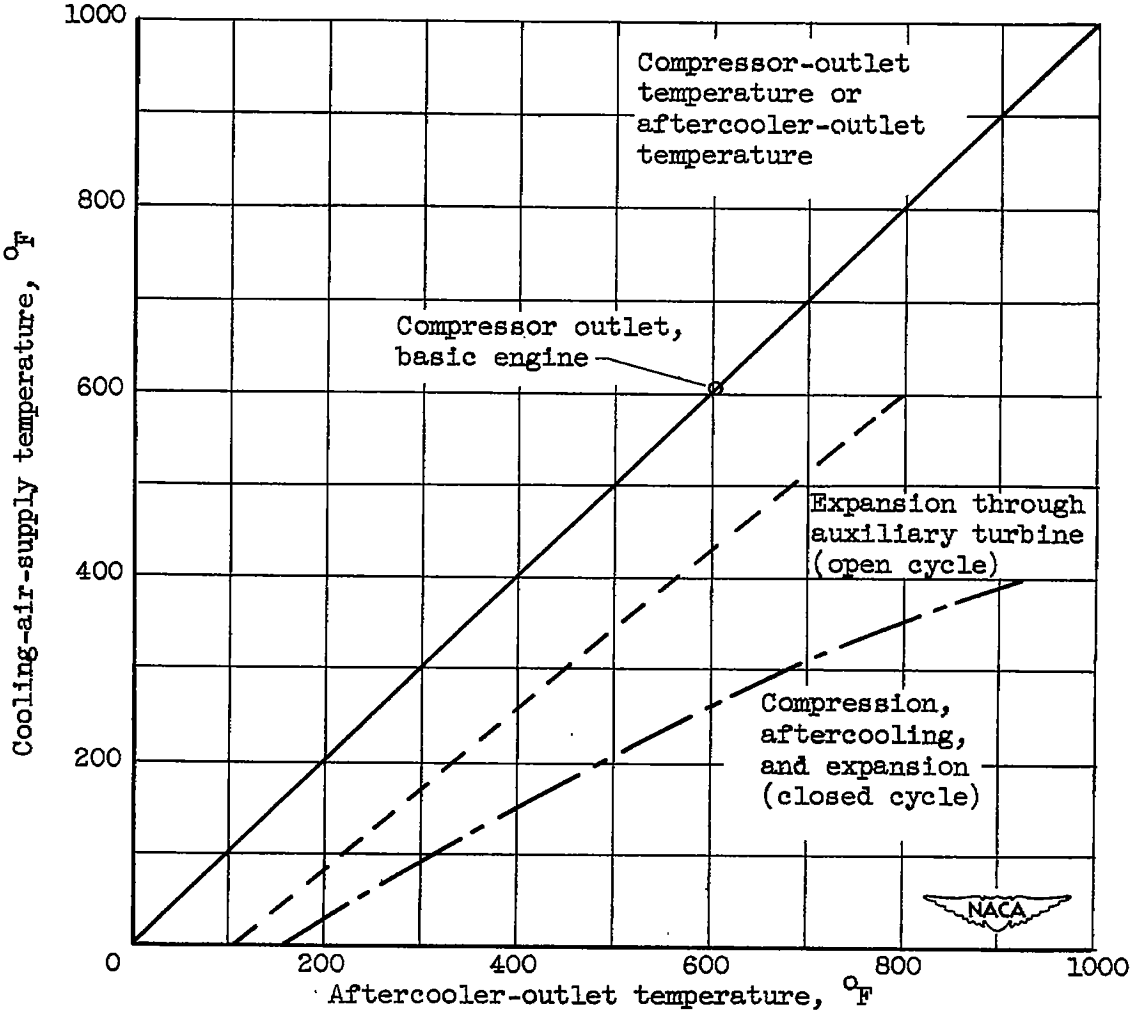


Figure 11. - Comparison of methods for refrigerating blade cooling air bled from engine compressor. Altitude, 50,000 feet; flight Mach number, 1.8; auxiliary compressor and turbine efficiencies, 0.85; final coolant pressure, 13.55 pounds per square inch absolute; pressure drop through aftercooler neglected.

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